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**FATIGUE AND FRACTURE BRANCH--A COMPENDIUM OF
RECENTLY COMPLETED AND ON-GOING RESEARCH PROJECTS**

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INTRODUCTION

The Fatigue and Fracture Branch of the Langley Research Center has a long tradition of providing basic research in support of fatigue and fracture problems in commercial and military aviation, as well as space transportation.

The Branch consists of NASA research scientists, and three Army (AVSCOM) scientists. In addition, five contract engineers work in the Branch utilizing NASA facilities.

In contrast to earlier years in which we frequently ran large numbers of tests to produce design oriented fatigue data, our mode of research today is oriented towards establishing and verifying fracture and damage models. In that mode, we now place equal emphasis on generating fracture knowledge for the materials development industry as well as characterizing the fracture behavior for the design industry. But less repetitive testing is required to test or verify models than was the case for generating statistically sound data for design use. At the same time, the development and expansion of fracture mechanics analysis has required a much larger computational effort, and many of our research projects are not directly accompanied by fracture experiments. In some respects that research orientation reflects that the aviation industry has to cope with a multitude of failure modes (e.g. composite impact) for which there are still no computational models.

Of all the materials problems we concentrate on fracture problems in structural materials, essentially airframe structures. Hence, we address both metals and composites. At the same time, recognizing that fatigue and fracture problems often originate at joints, one area of concentration is the fracture mechanics of bonded and bolted joints.

Our link to other research is through representation on a number of ASTM committees, through our extensive involvements with universities using research grants, as well as through industry contacts maintained either at technical meetings or in association with NASA project efforts such as the ACEE projects.

Our research results are published in journals or in NASA reports. While journal articles have the larger circulation, they are usually slow and restrictive in detail. Our NASA reports (TMs and TPs) are faster to produce, can contain more relevant details, but their distribution is generally low. To overcome this, we are making a major effort within the Branch to develop specifically directable mailing lists to target our reports to the appropriate research community.

This review of recently completed and on-going research projects is divided into three technical sections; metals, adhesive joints, and composites. Each identifiable project is described in a one-page write-up with a figure of results where applicable. Only the engineer responsible for the project is cited, even where the work is performed by a group.

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METALS

Our fracture mechanics program for structural metals is growing again after a ten-year period during which our primary emphasis was on composites. The new growth is controlled partly by the developments in powder-metallurgy products, but also by the new advances with ingot materials based on Aluminum-Lithium alloys.

For damage-tolerant design, modeling problems in ductile fracture and crack-growth under spectrum loading are primary issues with significant pay-off. Those are the long-standing problems for fracture mechanics in this Branch.

The development of more fatigue-resistant powder metallurgy products has required an additional research direction, the application of fracture mechanics to very small cracks. That work area is now growing, and is in a problem and method identification phase, which is largely experimental.

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CRACK CLOSURE AT NEAR-THRESHOLD FATIGUE CRACK GROWTH RATE

Project Engineer: E. P. Phillips

Tests have shown that cracks in metals will not grow due to cyclic loading unless the loads are high enough to cause the stress-intensity factor range (ΔK) to exceed a critical (threshold) value (ΔK_{th}). The threshold value observed is not only a function of the test material but also of several test variables, the most significant of which appears to be stress ratio and environment. The test method used to evaluate the threshold value may also affect the results. Since Elber introduced the concept of crack closure into the analysis of fatigue crack growth, P. K. Liaw of Westinghouse and several others have suggested that if crack closure is accounted for in threshold tests, a single "effective" stress-intensity factor range (ΔK_{eff}) may be found for all stress ratios. R. O. Ritchie at MIT has suggested that, in some cases, the variation in threshold values observed for different environments may also be explained by crack closure.

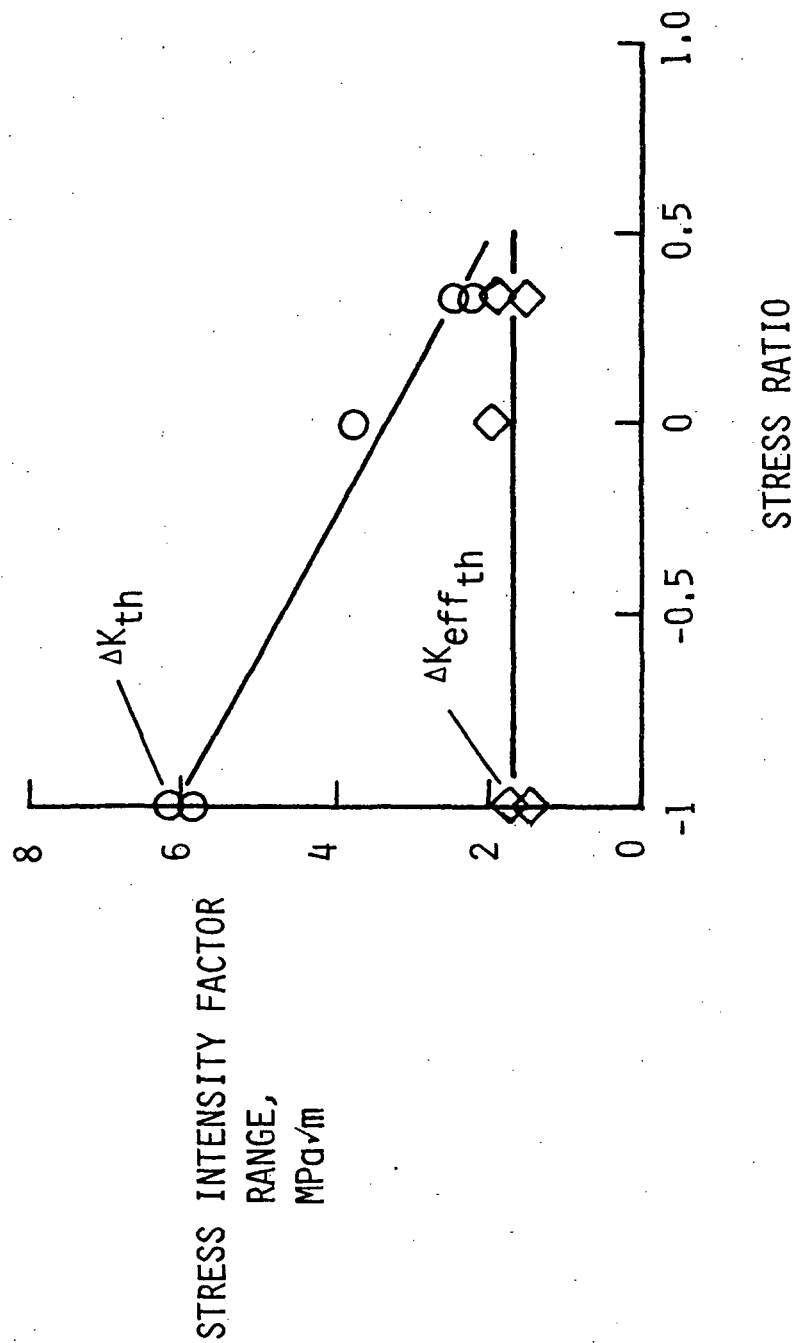
The long-range goals of our research in this area are to achieve a better understanding of (1) the contribution of crack closure to the development of fatigue crack growth thresholds in metals and (2) the mechanisms that produce crack closure at near-threshold rates. The potential applications of this research lie in the development of improved life prediction models for spectrum loading and in the development of alloys with higher growth thresholds.

Our short-range goal is to generate near-threshold rate data, define thresholds, and measure crack-opening loads for 2024-T3 aluminum sheet at several stress ratios. The threshold data are being generated by the load-shedding, constant stress ratio method. Crack-opening loads are being evaluated from load/COD records. The figure shows the variation of ΔK_{th} and $\Delta K_{eff_{th}}$ with stress ratio in our tests. Clearly, the value of ΔK_{th} changes substantially with stress ratio, but the value of $\Delta K_{eff_{th}}$ is essentially unaffected by stress ratio. This indicates that crack closure is playing a significant role in the development of the threshold in this material.

Because there is some doubt about the validity of the load-shedding test method, thresholds will also be determined in constant K_{max} , increasing stress ratio tests. Tests on other aluminum alloys will follow the above work to obtain a broad-base correlation of threshold performance with closure behavior.

VARIATION OF ΔK_{th} AND $\Delta K_{eff th}$ WITH STRESS RATIO

MATERIAL: 2024-T3 Al ALLOY



MEASUREMENT OF CRITICAL CRACK-TIP PROCESSES ASSOCIATED WITH FATIGUE CRACK GROWTH UNDER VARIABLE-AMPLITUDE LOADING

Project Engineer: E. P. Phillips

This is a contractual effort being performed by S. J. Hudak, D. L. Davidson, and K. S. Chan at Southwest Research Institute (SWRI). The work is currently in the second year of a two-year contract. Results of the first year's work are reported in a NASA Contractor Report.

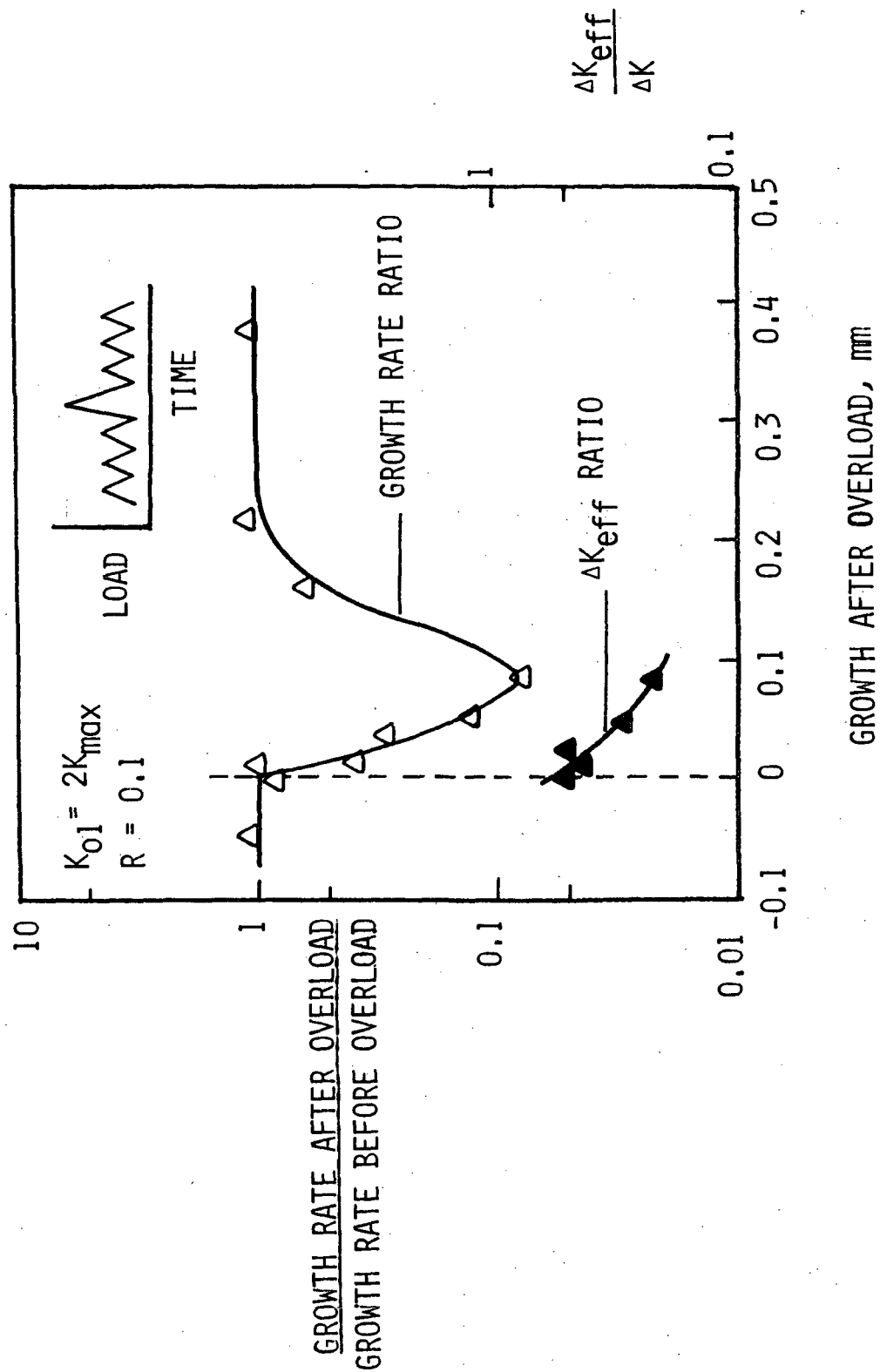
Growth of a fatigue crack in metals can be retarded or even stopped by the application of a high load. For structures subjected to a spectrum of loads, knowledge of the crack growth retardation following high loads is required to make accurate life predictions. Because crack closure is believed to control the crack growth retardation, efforts are being made to measure crack closure and to incorporate this information into improved life prediction models. Most evaluations of crack closure have been made using crack-opening-displacement (COD) measurements taken at a location remote from the crack tip. The unique feature of the SWRI work is the evaluation of crack-opening loads from measurements of displacements that occur very near the crack tip. This is accomplished by placing a cracked specimen inside a scanning electron microscope (SEM) and photographing the crack-tip area while the specimen is loaded in steps through the fatigue load cycle.

In tests completed by SWRI, growth rates and opening loads have been measured when a single overload was inserted into an otherwise constant-amplitude loading. Results from a typical test are shown in the figure. The open symbols represent crack growth rates and the closed symbols represent ΔK_{eff} values measured before and after the overload. The ΔK_{eff} values reflect the measurements of opening load since $\Delta K_{eff} = K_{max} = K_{open}$. As can be seen in the figure, the crack growth rate decreased dramatically just after the overload. The ΔK_{eff} values show the trend that would be expected if closure was controlling the retardation process.

In the second year of this effort, work will focus on determining the relative contributions to crack closure from (1) plasticity left in the wake of the advancing crack and (2) crack-tip residual stresses.

MEASURED GROWTH RATES AND ΔK_{eff} AFTER OVERLOAD

MATERIAL: 7091-T7E69 Al ALLOY



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CRACK-TIP-OPENING DISPLACEMENT RESISTANCE (V_R) CURVE METHOD

Project Engineer: J. C. Newman, Jr.

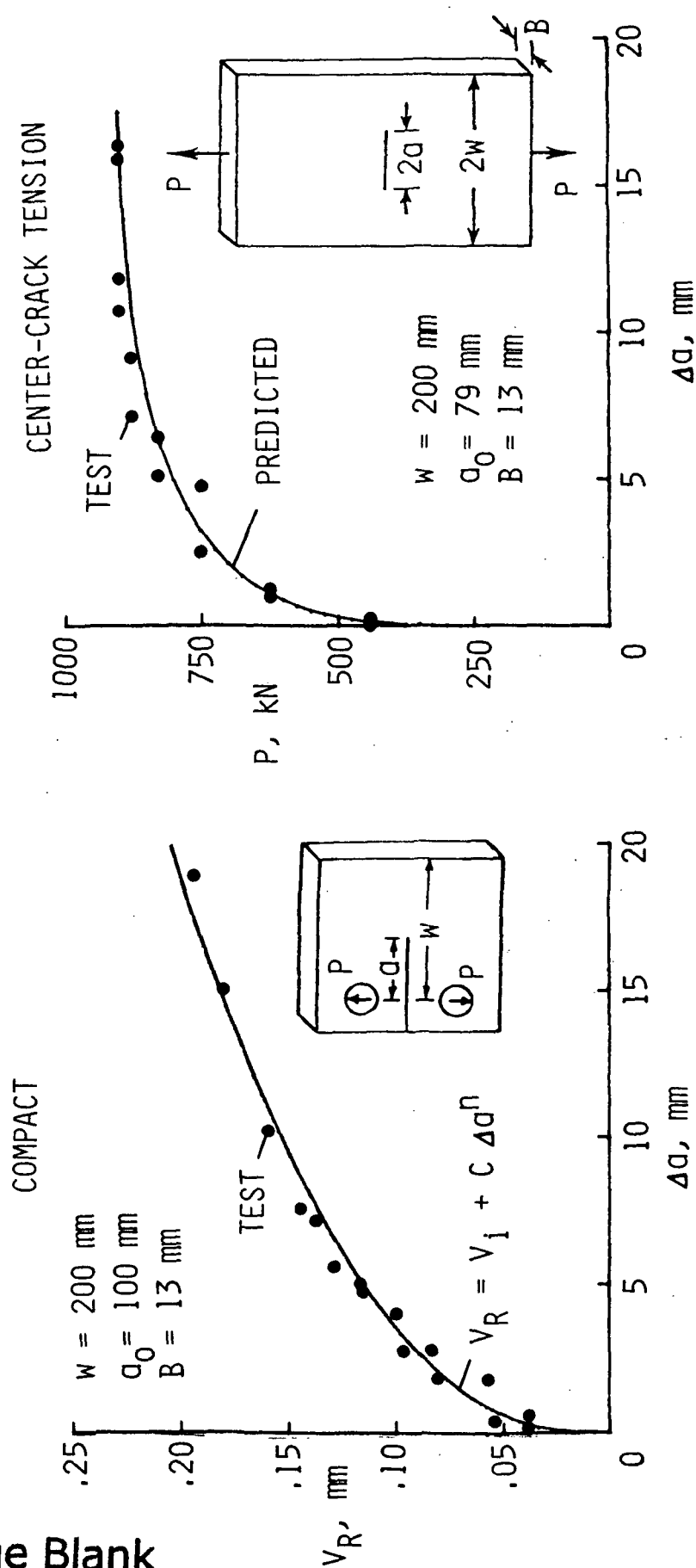
The resistance curve methods, such as K_R , J_R and V_R , are used to characterize the fracture behavior of materials. These concepts are used to predict stable crack growth and instability (maximum failure loads) of cracked structural components. The K_R -curve method (McCabe) is based on stress-intensity factor analyses and, consequently, applies only for materials with small plasticity. The J_R -curve method (Landes), on the other hand, is able to account for moderate to large plasticity at the crack tip, but the relation between J_R , crack length, specimen type, and tensile properties usually require a finite-element analysis. The V_R -curve method (Newman) is quite similar to the J_R -curve method except that the "crack drive" is written in terms of displacement instead of the J-integral. A relationship between the crack-tip-opening displacement (V_R), crack length, specimen type, and tensile properties has been derived from the Dugdale model. The Dugdale model solutions are easily obtained from superposition of two elastic crack problems. Consequently, the V_R -curve method can be applied to any crack configuration for which these two elastic solutions have been obtained.

The enclosed figure demonstrates the V_R -curve concept. Stable crack growth data (load against physical crack extension, Δa) on a large compact specimen made of 2024-T351 aluminum alloy was used to calculate the V_R data (symbols) from a newly-developed Dugdale model solution (Mall and Newman). For convenience, an equation was fitted to these data. The values of V_i , C , and n in the equation were assumed to be material constants dependent only on specimen thickness and test conditions (such as temperature and load rate). The V_R -curve was then used to predict the load-crack extension behavior on a center-crack tension specimen made of the same material. Predicted results, shown by the curve, agreed well with the experimental data (symbols). These results demonstrate the uniqueness of the V_R -curve and its ability to predict initiation, stable crack growth, and instability.

The application of the V_R -curve concept depends upon whether the Dugdale model solution for the particular crack configuration of interest is known. Efforts are underway to develop a Dugdale model analysis for any two-dimensional cracked plate. Because the analysis is elastic, a cost-effective computer program designated "FRACT" can be developed. This program will allow the user to specify the shape and loading on the cracked plate of interest. With the V_R -curve as input, the applied load and amount of crack extension will be output.

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EXPERIMENTAL CRACK-TIP DISPLACEMENT (V_R) RESISTANCE CURVE AND PREDICTED LOAD AGAINST CRACK EXTENSION FOR 2024-T351 ALUMINUM ALLOY



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SHORT CRACK GROWTH AND MICROSTRUCTURE

Project Engineer: J. C. Newman, Jr.

Growth of a crack from a small flaw (~ 0.01 mm) to one approximately 1 mm in length consumes a major part of the component's fatigue life in many engineering structures. Several investigators have reported that small cracks can grow at stress intensity levels below the threshold for long cracks and will grow at rates faster than long cracks above the threshold value. Lankford studied small cracks initiating on the sheet surface of unnotched 7075-T6, while Taylor and Knott used nickel aluminum bronze where casting defects provided crack initiation sites.

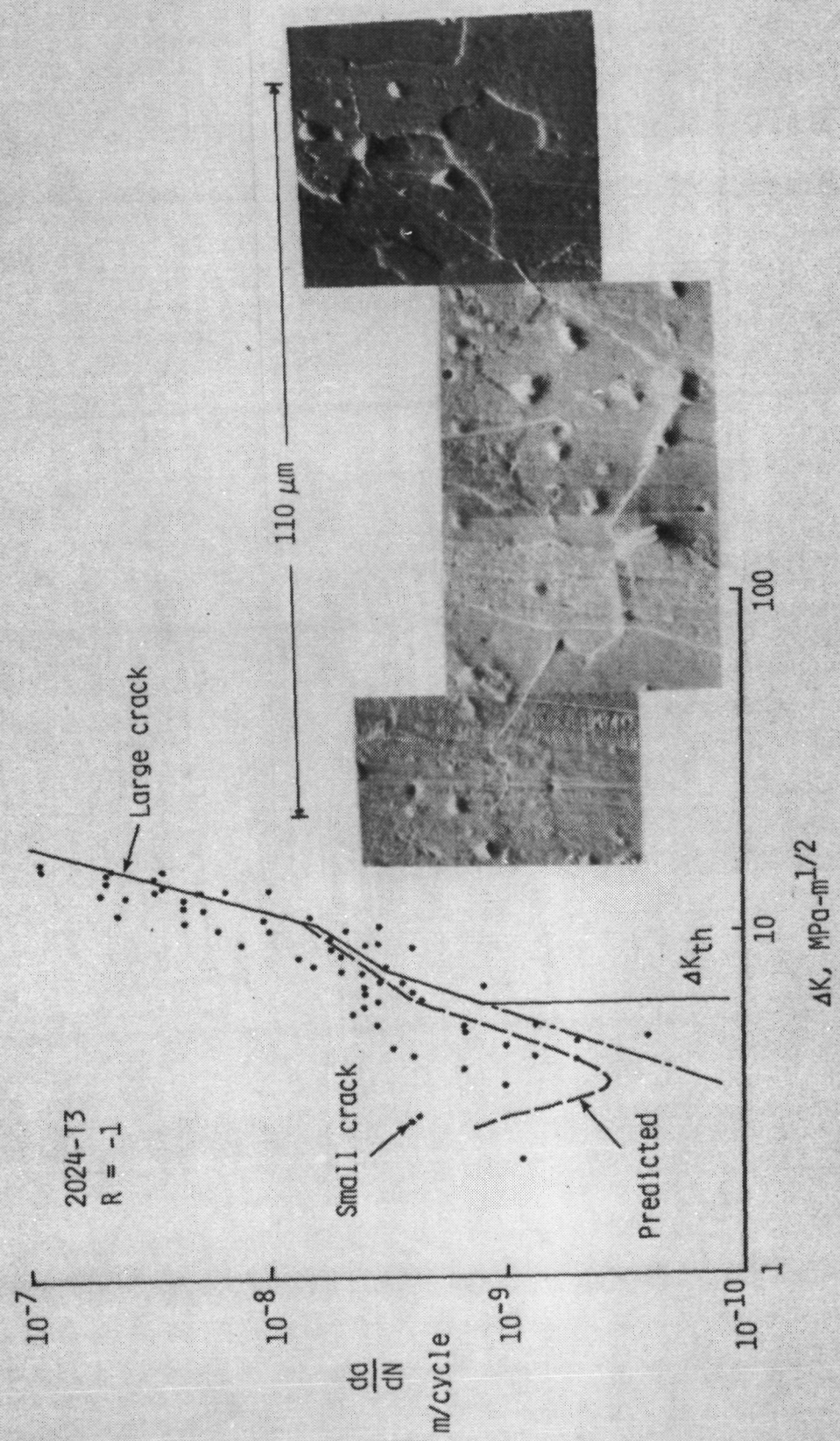
We have an experimental program to study the small crack effect in 2024-T3 aluminum alloy. The goals of that program are to:

- o Produce crack growth rate data both above and below the long crack threshold stress intensity range to verify the small crack effect.
- o Establish the dependence on R and maximum stress.
- o Characterize the initial flaw and determine the shape of the surface crack as a function of its size.
- o Evaluate crack surface topography as affected by microstructure (grains and inclusion particle distribution).
- o Use the above results in conjunction with ongoing analytical efforts to evaluate the level to which fracture mechanics can be used to predict the small crack effect.
- o Extend the continuum and noncontinuum findings to other and next generation alloy systems.

Testing of a semicircular edge notched specimens has been completed at one stress level for $R = -1$. Crack size was monitored using plastic replicas made at intervals during the test and examined in the scanning electron microscope. Results are shown in the figure. The cracks initiated at inclusion particle clusters and propagated in a transgranular mode producing fracture surface facets of the order of the grain size. Data were obtained on cracks from 7 μ m in length until the point when breakthrough occurred at one or both notch edges. Small cracks grew at accelerated rates both above and below the long crack threshold. The data blend with long crack data for $\Delta K \approx 10 \text{ MPa}\sqrt{\text{m}}$.

This program will be expanded to include the peak aged and overaged conditions for this alloy, and next generation powder metallurgy and/or aluminum alloys. A commitment has been made to participate in AGARD Short Crack Round Robin, a similar test program on 2024-T3 which additionally will include spectrum loading tests.

CRACK-GROWTH RATES FOR SMALL AND LARGE CRACKS



A CRACK-OPENING STRESS EQUATION FOR FATIGUE CRACK GROWTH

Project Engineer: J. C. Newman, Jr.

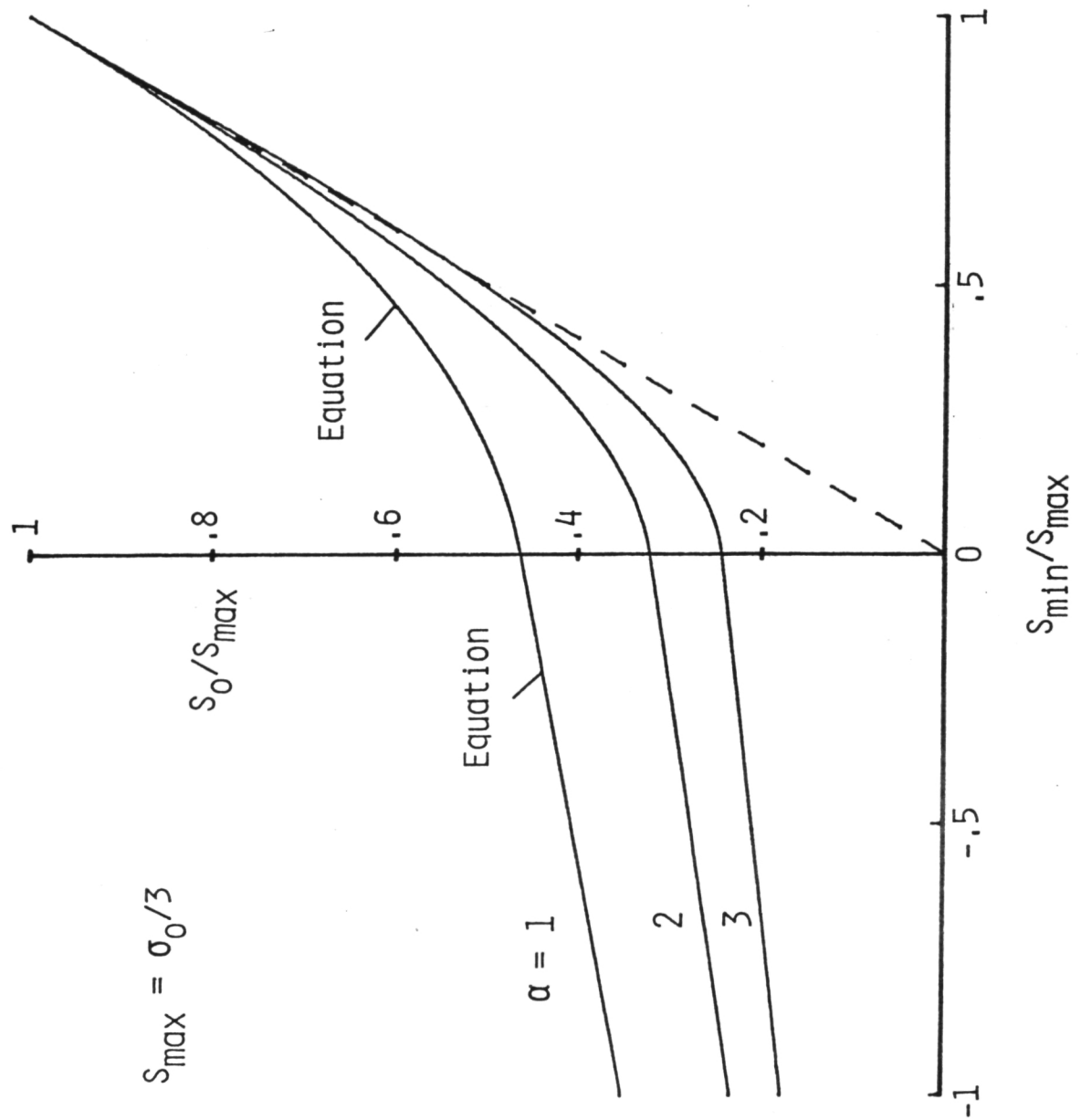
Experiments on metals have shown that fatigue cracks, under constant- and variable-amplitude loading, remain closed during part of the load cycle (Elber). There have been several attempts to develop simple analytical models of crack closure to calculate crack-opening stresses (S_o). Most of these models were based on a concept like the Dugdale model, but were modified to leave plastically-deformed material in the wake of the advancing crack. These two-dimensional models have shown that S_o is a function of stress ratio and stress level (S_{max}). Crack-opening stresses are also a function of specimen thickness (or three-dimensional constraint).

A general crack-opening stress equation for constant-amplitude loading is presented here. The equation is a function of stress ratio (R), stress level (S_{max}), and three-dimensional constraint. The effects of three-dimensional constraint have been simulated in a two-dimensional closure model by using a "constraint" factor (α) on tensile yielding; that is, the material yields when the stress is $\alpha\sigma_o$. The material is assumed to yield in compression when the stress is $-\sigma_o$. The influence of constraint on calculated crack-opening stresses is shown in the enclosed figure. The normalized crack-opening stress is plotted against stress ratio for $S_{max}/S_o = \sigma_o/3$. The curves, calculated from the equation, show how S_o/S_{max} varies as conditions change from simulated plane stress ($\alpha = 1$) to simulated plane strain ($\alpha = 3$).

For a material of given thickness and loading, the precise effect of three-dimensional constraint on crack-opening stresses is, as yet, unknown. However, at high values ΔK (plastic-zone sizes greater than plate thickness), plane-stress conditions ($\alpha = 1$) should exist. But at low values of ΔK (plastic-zone sizes less than about 10 percent of plate thickness), plane-strain conditions ($\alpha = 3$) should prevail. The crack-opening stress equation is being used to correlate crack-growth rate data on various materials and thicknesses under constant-amplitude loading. A closure model that accounts for changing constraint (from plane strain to plane stress) has recently been used to successfully predict crack growth under spectrum loading.

Efforts are underway to measure crack-opening stresses during the transition from plane-strain to plane-stress conditions, near threshold, and under other conditions (Phillips). These results will help the analyst verify the analytical calculations and, consequently, improve the damage-tolerant design procedures.

CRACK OPENING STRESS AS A FUNCTION OF STRESS RATIO AND CONSTRAINT FACTOR



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MOST COMMON CRACKS IN AEROSPACE STRUCTURES

Project Engineer: J. C. Newman

Corner cracks and through-cracks at holes are among the most common flaws in aircraft structures. The photograph in the enclosed figure shows a corner crack at a hole in a plank on a fractured wing structure. Accurate stress analyses of such configurations are needed to reliably predict crack growth rates and fracture strengths, and to establish inspection intervals so that failures like this one can be avoided.

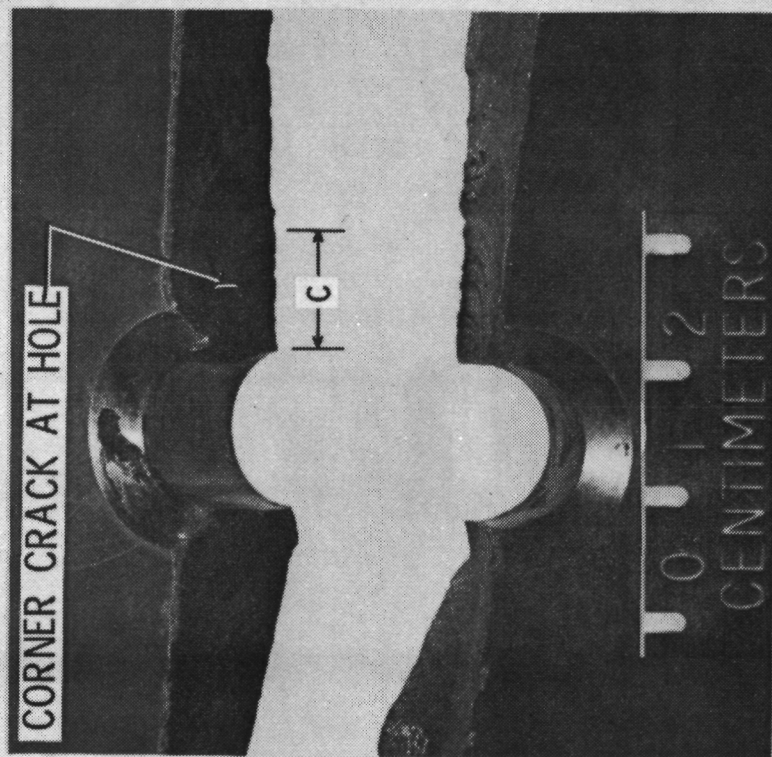
A three-dimensional finite-element analysis was conducted by Raju and Newman to determine stress-intensity factors for quarter-elliptical corner cracks at the edge of a hole under various loading. These stress-intensity factor solutions are used to predict the number of load cycles required to grow a crack from a small defect at the edge of a hole to failure. The enclosed figure shows a comparison between experimental and predicted crack length against cycles for a corner crack growing from a circular hole in a 7075-T651 aluminum alloy specimen subjected to constant amplitude tensile loading. The predicted crack growth life was in good agreement with the experimental data.

Equations for stress-intensity factors for other three-dimensional crack configurations (such as an embedded elliptical crack, a quarter-elliptical corner crack, a semi-elliptical surface crack, and surface and corner-cracks at a hole) subjected to tensile and bending loads have also been developed by Newman and Raju. Many of these equations have been successfully applied to predict crack growth of these crack configurations.

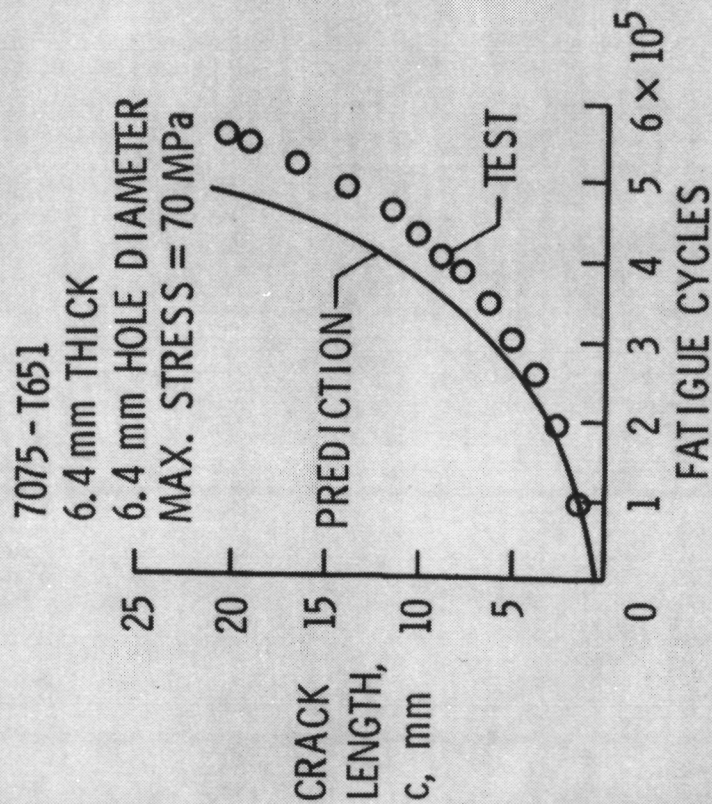
Efforts are underway to analyze other three-dimensional crack configurations, such as surface-cracks, in rods, cylinders, and lugs by the finite-element method. Equations for the stress-intensity factors for these configurations will be developed.

A stress-intensity factor compendium of three-dimensional crack configurations will help engineers design structural components that are safe, economical, and damage tolerant.

WING FAILURE DUE TO CORNER CRACK AT HOLE AND EXPERIMENTAL VERIFICATION OF ANALYSIS



FAILURE OF PLANK IN WING STRUCTURE



PREDICTING FATIGUE CRACK GROWTH USING A CLOSURE MODEL



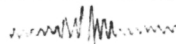
Project Engineer: Herbert Leybold

A program for predicting fatigue crack growth under both constant and variable amplitude loading has been developed by Newman. The state of stress was approximated with a constraint factor that remains constant regardless of the state of stress. In the limit it has a value of about 2.7 for plane strain and 1.0 for plane stress conditions. Using a constant constraint factor the program over predicts life for low stress conditions and under predicts life for high stress conditions. In order to predict life more accurately, the program was modified to include a constraint factor that could vary with changing states of stress. To do this, the transition region between plane strain and plane stress needs to be accurately defined. This can only be done with extensive constant amplitude data. The success of using the variable constraint factor depends on how accurately the limits of the transition region can be defined. Where extensive constant amplitude data was available the variable constraint factor provided considerably better life predictions than with the constant constraint factor.

To check the accuracy of the model, predictions were made on an extensive set of test data reported by Schijve. A comparison of predicted and test lives is shown in the table. On the average the program underpredicted the test results by about 20 percent. In order to determine whether the type of spectrum had any effect on the predictions, other loading spectrums such as TWIST, MINITWIST, FALLSTAFF, and a GAUSSIAN spectrum will be evaluated.

Ways of improving the efficiency of the program are always being evaluated. It was determined that the precracking phase could be greatly shortened by considerably increasing the size of the crack tip elements. This was done without significantly affecting the accuracy of the predictions.

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	GTAC	GUST	LOADS	N /N predicted Test	
	S * min	S * a ,max.	S * a ,min.	7075-T6	2024-T3
	-1.4	12.1	1.1	.42	.79
		7.7		.67	
		6.6		1.10	.68
		5.5		1.10	
		4.4		1.11	.96
	-3.4	8.8	1.1	.48(.48)	.62
		7.7		.61(.56)	.55(.55)
		6.6		.74(.85)	.94(.87)
		5.5		.77	.91
		4.4		.90	1.00
		6.6	2.2	.75	.78
		7.7	3.3	.60	.60(.58)
		6.6		.94(1.06)	1.01(.99)
		1 gust	load/ft.		
		S a,max =	6.6	1.16	1.07
	Omitted	6.6	1.1	1.16	1.07
				- X=0.81 σ=.25	- X=0.80 σ=.18

2
*kg/mm

THE MULTI-PARAMETER YIELD ZONE LOAD INTERACTION CRACK GROWTH MODEL

Project Engineer: W. S. Johnson

Most aerospace structures are subjected to random loadings. These random loads do not accumulate crack growth in a linear fashion. Indeed, load interactions occur which may cause crack growth to slow down or speed up significantly. Therefore, models which can predict these load interactions are required for safe aircraft design.

The multi-parameter yield zone (MPYZ) model is based upon a residual-stress-at-the-crack-tip concept. The model requires empirical phenomenological constants to correctly describe a particular material. The model accounts for crack growth acceleration, retardation, multiple overloads, and underload effects. The model has been demonstrated in several ASTM round robins to be one of the best available.

The MPYZ model is currently used at General Dynamics/Fort Worth, Grumman Aerospace Corporation, Hughes Tool Company, University of Dayton Research Institute, and Boeing/Wichita.

No future work is currently planned.

Publication:

Johnson, W. S.: Multi-Parameter Yield Zone Model for Predicting Spectrum Crack Growth. Methods and Models for Predicting Fatigue Crack Growth Under Random Loading, ASTM STP 748, 1981, pp. 85-102.

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CONSTRAINT EFFECTS ON FATIGUE CRACK PROPAGATION

Project Engineer: W. S. Johnson

In most fatigue crack propagation predictions, one of the primary sources of error stems from differences in propagation between plane stress and plane strain. These differences are now being addressed systematically.

The purpose of this grant with Ben Hillberry of Purdue University is to conduct an experimental investigation of the constraint effects due to thickness on the fatigue crack growth rate of an aluminum alloy. The test program consisted of a series of constant-amplitude fatigue tests at two stress ratios, 0.05 and 0.75, and a series of fatigue tests with a single overload using the same stress ratio. Three different specimen thicknesses were tested: 0.08, 0.17, and 0.25 inch. For each thickness, the loading program was selected such that the growth progressed from a state of plane strain to a state of plane stress. As already reported by Hillberry, the overload ratio was the same for most tests, and each thickness was tested under the same stress-intensity condition to determine the influence of the constraint due to thickness on the crack growth following the overload.

These data will be incorporated into the crack closure and CGR-LaRC programs at Langley to enable more accurate crack growth predictions.

ADHESIVE JOINTS

Bonding of entire structures with the aim of reducing stress concentrations has great appeal but has never been entirely successful in aircraft structures. The past lack of understanding of bonding's own stress concentrations is probably the single biggest reason for that. In recent years the developments in fracture mechanics have provided sophisticated new methodology to reanalyze bonded joints.

The work in this Branch is aimed at translating more of the existing fracture mechanics technology to the analysis and design of bonded joints, to improve the reliability of bonded joint designs.

CYCLIC DEBONDING OF ADHESIVELY BONDED COMPOSITES

Project Engineer: W. S. Johnson

Composite Structures lose 30-40 percent of their load carrying capability when mechanical fastener holes are introduced. Adhesive bonding may be an attractive alternative. The purpose of this research is to examine the possible failure modes of adhesively bonded graphite-epoxy, characterize the cyclic debond behavior, and suggest a potential design approach.

Two adhesive systems have been examined: EC-3445 (a 250°F cure paste adhesive) and FM-300 (a 350°F cure mat adhesive). It was found that the cracked-lap-shear (CLS) specimens with 0° plies on the surface resulted in stable debonding in the adhesive, with no delamination in the composite adherends. Very respectable debonding as a function of total strain energy release rate, G_T , was obtained for both adhesive systems. The total strain energy release rate approach was verified by using a tapered CLS specimen to predict debond initiation based on a G_T threshold. Additional testing showed that the debond growth rate data was the same for a double-cantilever-beam (DCB) specimen as it was for the CLS specimen based on G_T . The adhesives systems used are directly applicable to the Army's ACAP and the Navy's F-18 fighter.

Future research includes looking at a wider range of GI/GII ratio data and other adhesive systems.

Publications

- o Mall, S.; Johnson, W. S.; and Everett, R. A., Jr.: "Cyclic Debonding of Adhesively Bonded Composites" NASA TM 84577, November 1982.
- o Johnson, W. S; and Mall, S: "A Fracture Mechanics Approach for Designing Adhesively Bonded Joints," NASA TM 85694, September 1983.
- o Johnson, W. S.; Mall, S., Everett, R. A. Jr.: "Adhesive Bond Strength: Static vs. Fatigue, SESA, June 1984.
- o Mall, S., and Johnson, W. S.: "Characterization of Mode I and Mixed-Mode Cyclic Debonding of Adhesively Bonded Composites, ASTM April 1984.

Upcoming research is concerned with verifying that the total strain energy release rate can predict both failure and debond growth rate over the complete range of normal peel-to-shear ratios. A contract with E. Ripling of Materials Research Labs is already underway to study the influence of combined peel and transverse shear modes for the same adhesives.

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EXPLORATORY STUDY OF FAILURE MODES IN REALISTIC BONDED COMPOSITE STRUCTURES

Project Engineer: R. A. Everett, Jr.

An exploratory study is being conducted to determine the possible failure modes that occur in "realistic" adhesively bonded joints. Failure modes are being determined for both monotonic static loading and constant-amplitude cyclic loading. For this study, three different joint configurations were designed consisting of titanium and graphite/epoxy adherends bonded with a high-strength structural adhesive. A secondary objective of this study is to compare the static strength of the joints to the predicted design strength. Currently, two joint configurations have been tested to failure statically, and one configuration has been tested to failure under cyclic loading. Although the joints were designed to fail in the adhesive, both joint configurations failed by delamination in the graphite/epoxy adherends for both the static and cyclic tests. The static strengths of the joints that have been tested were at least 30 percent greater than the predicted strengths. Since the failure modes were by composite delamination, the failure strength of the adhesive is even greater than 30 percent more than the predicted strength.

One possible reason for the joint not failing in the adhesive is the use of the single-lap-shear test to obtain the maximum shear strength of the adhesive. One of the principal design parameters for predicting failure in this analysis was the adhesive shear strength. Current analysis of the single-lap joint shows that a high concentration (possible singularity) of normal stresses exists at the end of the lap, implying that the single-lap joint fails by a combination of peel and shear stresses; thus, the adhesive's actual shear strength is greater than that determined from the single-lap-shear specimen. Because of the possible singularity at the end of the lap, a fracture mechanics analysis, using the critical strain-energy-release rate as a design parameter, could explain the apparent conservative design.

Because we are more concerned with debond failure mechanisms than with conservatism, we will review the design in terms of fracture mechanics analysis to see if we can predict the high joint strength. We will also modify (weaken) the joints until both static and fatigue failures occur in the adhesive, in order to be able to study the important failure mechanisms.

REPEATABILITY OF MIXED-MODE ADHESIVE DEBONDING

Project Engineer: R. A. Everett, Jr.

One of the factors that has delayed the widespread use of adhesively bonded structures, especially in primary aircraft structures, is the question of joint reliability. Even though reliable fabrication methods have been developed, there will always be variations in these methods from manufacturer to manufacturer, as well as differences that arise from other sources such as material variability. If the service life of bonded structures is going to be predicted with any satisfactory accuracy using analytical techniques such as fracture mechanics, the repeatability of experimental data must be within an acceptable scatter band. The purpose of this study was to obtain data on adhesive bond repeatability by comparing debond growth data from specimens made by two different manufacturers and tested in different laboratories.

Debond growth data were obtained from constant-amplitude fatigue tests that were conducted on cracked-lap-shear specimens consisting of aluminum adherends bonded with a high-strength structural adhesive. Critical values of strain-energy-release rate were also determined from specimens that were monotonically loaded to failure. The results of this study showed that the debond growth rates for the two sets of specimens varied by a factor of 2 to 7, which is similar to that observed in fatigue crack growth in metals. The tests also showed that cyclic debonding occurred at strain-energy-release rates more than an order of magnitude below the critical values.

Since fracture mechanics techniques have been used successfully in describing fatigue crack growth in metals, and the debond growth rates from this study exhibit a scatter similar to that in metals, the results of this study give confidence that fracture mechanics can be used to describe similar behavior in adhesively bonded structures. This study has further shown that the design of adhesively bonded structures should consider the fatigue failure mode also, since cyclic debonding can occur at strain-energy-release rates that are an order of magnitude below the critical values.

Publication:

Everett, R. A., Jr.; and Johnson, W. S.: Repeatability of Mixed-Mode Adhesive Debonding. NASA TM-85753, February 1984.

STRESS ANALYSIS OF ADHESIVELY BONDED STRUCTURES

Project Engineer: R. A. Everett, Jr.

Because bonded structures can be light, cheap, and strong, aerospace and automotive companies are considering bonding airframes and car bodies. One drawback, though, that hampers immediate widespread application is apprehension about the durability of the bonded joints. This is because the stresses and deflections in bonded joints are nonlinear and, until recently, very imperfectly understood.

To gain insight into bondline behavior, a finite-element analysis called GAMNAS (Geometric and Material Nonlinear Analysis of Bonded Structures) has been developed. Currently, the analysis can account for nonlinearity arising from large rotations (geometric nonlinearity) and nonlinear material behavior. The program also calculates strain-energy-release rate at the debond front.

This analysis has been used to study the influence of adhesive properties on the strain-energy-release rate, G , which has been related to failures in bonded joints. Currently, analysis has shown that G is more sensitive to the adhesive's shear modulus than to either Poisson's ratio or the bondline thickness. Further analysis has shown that a linear geometric analysis of a joint, such as the cracked-lap-shear specimen, will show G to vary with debond length, whereas a nonlinear geometric analysis showed G to be relatively constant with debond length.

This analysis has been a useful analytical tool which is being used to design test specimens that are used to determine relationships between G and bonded joint durability.

Publications:

1. Dattaguru, B.; Everett, R. A., Jr.; Whitcomb, J. D.; and Johnson, W. S.: Geometrically Nonlinear Analysis of Adhesively Bonded Joints. Journal of Engineering Materials and Technology, vol. 106, no. 1, January 1984, pp. 59-65.
2. Mall, S.; Johnson, W. S.; and Everett, R. A., Jr.: Cyclic Debonding of Adhesively Bonded Composites. NASA TM-84577, November 1982.

METHODS FOR DETERMINATION OF ADHESIVE SHEAR MODULUS

Project Engineer: Carl E. Rucker

Adhesive properties are subject to considerable variation depending on handling and joint design. This study was designed to provide a systematic method to begin to compare measuring techniques for determination of the adhesive shear modulus in lapshear joints. Specimen design was selected in order to provide comparison with an existing data base but also to obtain reduced stress concentrations in the joint. Materials were also selected which have a strong data base. FM73M was selected as a prominent structural adhesive and a phosphoric acid anodize surface treatment was selected as a well documented cleaning system. Thick adherends of 0.075 thickness were fabricated by a major airframe contractor and a local contractor with limited bonding experience. The test specimens were evaluated using two non-destructive evaluation techniques as measures of the shear modulus. The specimens will subsequently be destructively tested and a third determination of the shear modulus will be made based on distortion of the joint while being loaded. Statistical comparisons will be made between shear modulus determination by the three techniques and by process variations where applicable.

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THE VISCOELASTIC CHARACTERIZATION AND LIFETIME PREDICTIONS OF STRUCTURAL ADHESIVES

Project Engineer: W. S. Johnson

The objective of this work by Hal Brinson of Virginia Tech is to develop a procedure to predict the failure of adhesive joints where service life must span 10 to 20 years using, as a basis, analytical projections or extrapolations from short-time test data. Most polymers are subject to viscoelastic behavior. Such behavior can result in rupture over a period of time if the temperature and stress level are sufficient. From a design point of view, we wish to determine this envelope of time, temperature, and stress that may lead to rupture such that it can be avoided in design.

To date, FM-300 and FM-73 have been creep tested in the neat condition. A finite-element program has been used to predict stress-strain response of the adhesive in a single-lap-shear specimen using the neat adhesive data. The predictions compared favorably with strains measured from an embedded strain gage in the joint. Some of these results have already been published.

Future work by Dan Post of Virginia Tech will include measuring displacements in the adhesive bondline of a thick-adherend specimen and a cracked-lap-shear specimen using the Moire fringe technique. These data will be used to assess Langley's geometric and material nonlinear finite-element program.

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CRACK PROBLEMS IN PLATES, SHELLS, AND BONDED MATERIALS

Project Engineer: W. S. Johnson

The purpose of this grant is to develop stress-intensity factor (SIF) solutions for various crack geometries in plates, shells, and bonded joints. Fazil Erdogan of Lehigh University has produced numerous SIF solutions over the years and is continuing his work under grant.

During this past year, SIF solutions were produced for pressurized cylindrical shells with fixed and stress-free ends that contain an axial part-through or through crack. Erdogan also produced a solution to an adhesively bonded cracked-lap-shear specimen using the Reissner plate theory. Many of Erdogan's closed-form solutions have been incorporated into stress solution handbooks and also have served as precision checks for finite-element model solutions. His many results are published in the open literature and in NASA Contractor Reports.

Planned work includes solutions for a finitely cracked layer of a multilayer laminate. Both the SIF within the cracked layer and at the interface parallel to the cracked layer will be examined. These solutions should be directly related to laminated metals and composites research.

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COMPOSITES

The great potential weight savings possible with resin-matrix composites have been the overwhelming driver for the large effort in composite fracture mechanics. Especially those problems peculiar to the laminated composites such as delamination from the edges and after impact are ideally analyzable with fracture mechanics, and are being extensively worked. Also, compression fracture is a problem peculiar to composites, in that no exact parallel exists in metal structures. The methodology for such phenomena often includes large post-buckling behavior and some progress is being made in work here.

The traditional tension fracture problem treated in metal structures, has not received as much attention in the design and applications, partly because the tendency to delaminate enhances the material fracture toughness. But the development of stronger and tougher resins may give the fracture problem its traditional importance. We, therefore, are developing fracture analysis methods for both thick and thin laminates, including hybrids and buffer-strip-protected laminates.

In order to contribute to the search for both better matrices and fibers, we are also formulating an extensive program for the development of micro-mechanics analyses.

IMPROVED TEST METHODS FOR MEASURING INTERLAMINAR FRACTURE TOUGHNESS OF COMPOSITES

Project Engineer: T. Kevin O'Brien

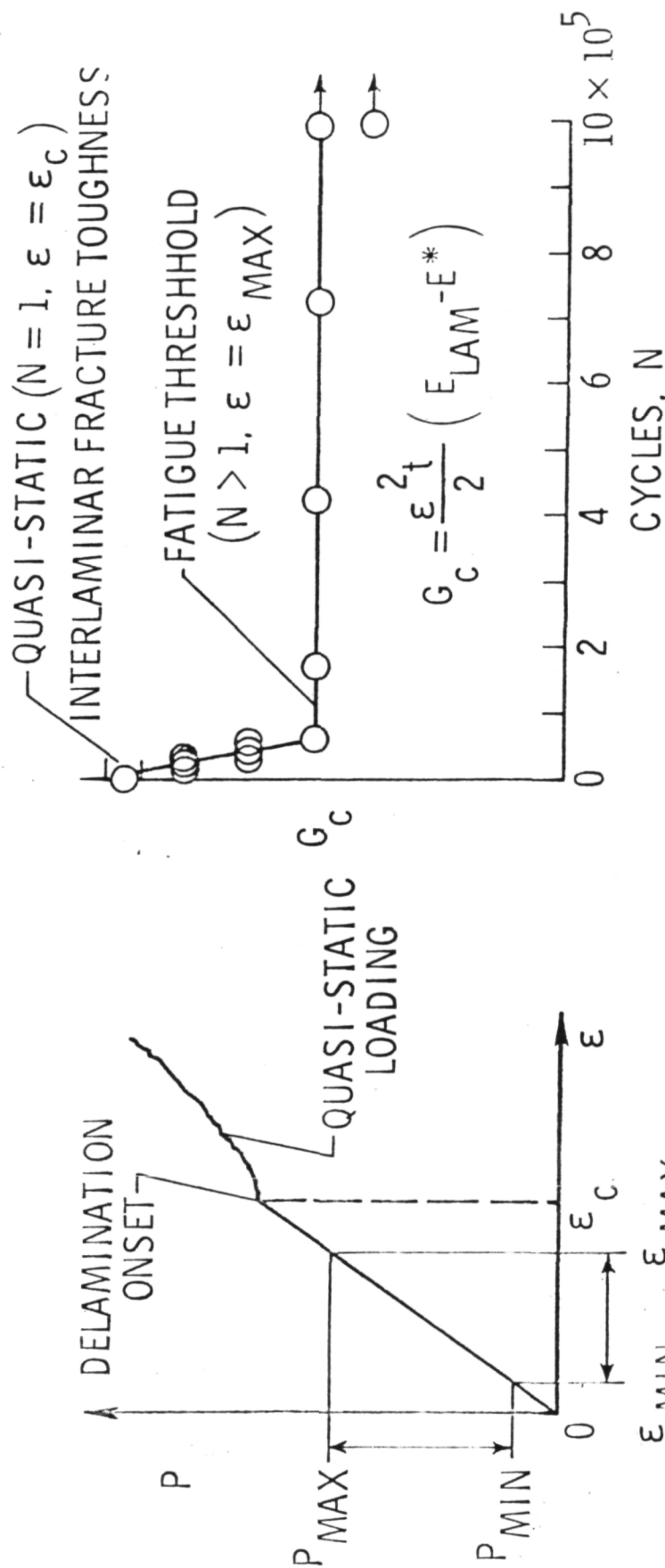
Fundamental studies of the mechanics of composite delamination starting from stress free edges has led to the development of the edge delamination tension test (EDT) for measuring composite interlaminar fracture toughness. Original experimental and analytical work conducted in-house led to the development of strain energy release rate solutions for edge delamination. Further refined finite element analysis of the various fracture modes inherent in the EDT test is being performed by the U. of Illinois (S. S. Wang). Some experimental work examining specimen geometry is being conducted at VPI&SU (D. H. Morris). Toughened-resin composites are being evaluated using the EDT test by Lockheed, Boeing, and Douglas under the ACEE project office "Key Technologies Contracts." Further work is being conducted in cooperation with AFWAL to evaluate the EDT test with Teflon inserts to act as crack starters for G_{IC} or G_c measurement. An ASTM round robin will soon be underway to evaluate the EDT test.

Fatigue delamination studies have shown that cyclic loading may cause extensive delamination in graphite composites, even for laminates that do not delaminate under static loads. Therefore, it is necessary to characterize delamination resistance in fatigue as well as in static loading. To this end, various EDT layups made with different resin matrices were cycled to strain levels below the delamination onset strain measured in the static tests. Delaminations formed at these lower cyclic strains after a certain number of cycles, N . As shown in the figure, strain-energy-release rate (G) values, calculated from maximum cyclic strains and plotted as a function of the number of cycles to delamination onset, dropped sharply and then reached a plateau tantamount to a threshold for delamination onset in fatigue. The fatigue delamination resistance of toughened-resin composites is being characterized using this technique at the U. of Wyoming (E. Odem).

- o O'Brien, T. K.: "Characterization of Delamination Onset and Growth in a Composite Laminate," in Damage in Composite Materials, ASTM STP 775, 1982, p. 140.
- o O'Brien, T. K., Johnston, N. J., Morris, D. H., and Simonds, R. A.: "A Simple Test for the Interlaminar Fracture Toughness of Composites," SAMPE Journal, vol. 18, no. 4, July/August 1982, p. 8.
- o O'Brien, T. K.: "Mixed-Mode Strain-Energy-Release-Rate Effects on Edge Delamination of Composites," in "The Effect of Defects in Composite Materials," ASTM STP 836, 1984. Also in NASA TM-84592, January 1983.

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CHARACTERIZING DELAMINATION RESISTANCE IN FATIGUE WITH EDGE DELAMINATION TEST



CYCLIC
LOADING

LAMINATE CONFIGURATIONS

$$\begin{aligned} & \left[\pm 35/0/90 \right]_s \\ & \left[\pm 45/0/90 \right]_s \\ & \left[\pm 30/\pm 30/90/\overline{90} \right]_s \end{aligned}$$

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FATIGUE OF UNNOTCHED COMPOSITE LAMINATES

Project Engineer: T. Kevin O'Brien

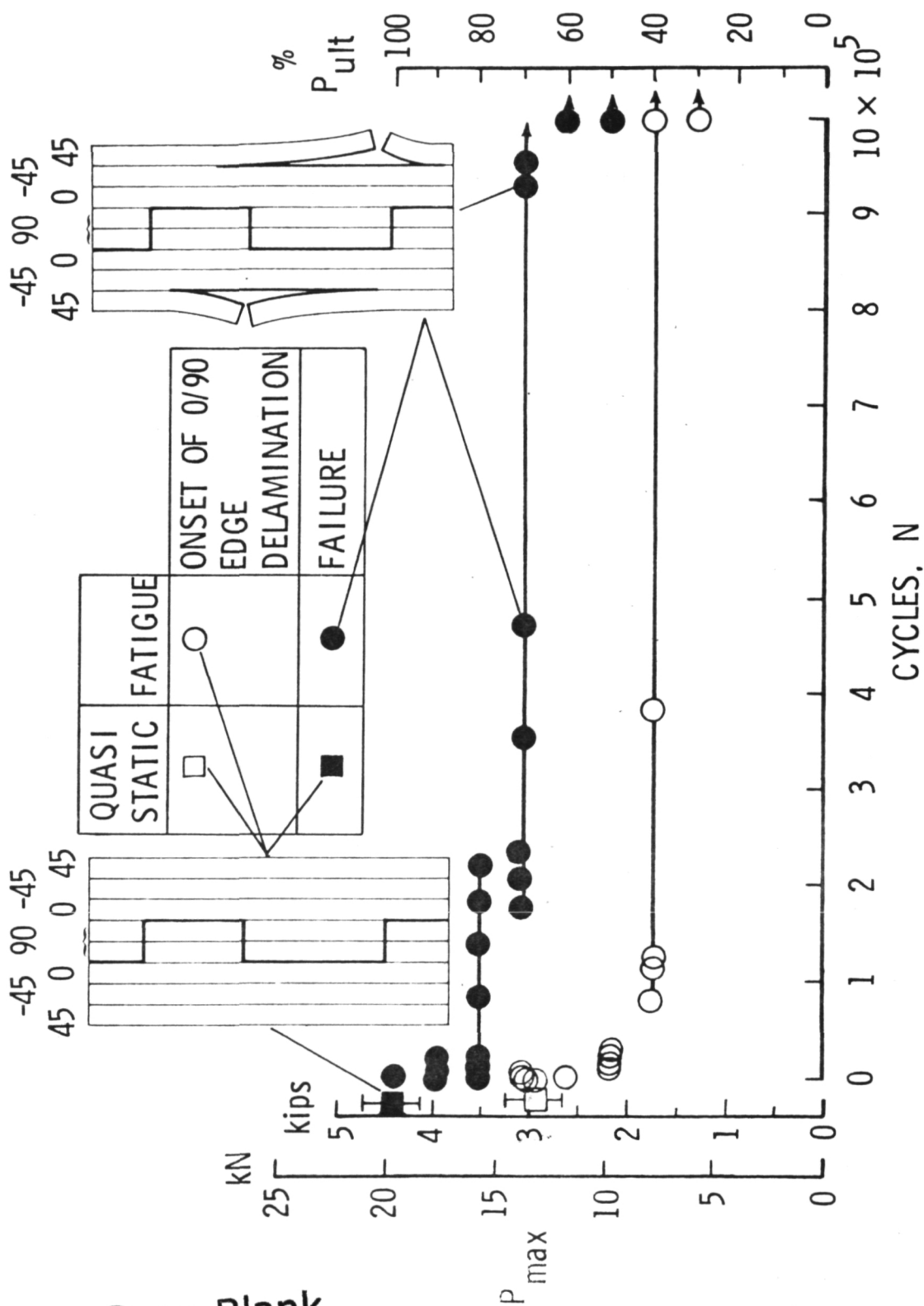
The influence of delamination (in-house) and matrix cracking (Lockheed California - Jim Ryder and Frank Crossman) on stiffness, strength, and fatigue life were evaluated. The objective was to develop a basic understanding of the influence of damage on mechanical properties for a wide variety of layups.

An in-house study of damage development during tension-tension fatigue loading of unnotched $[\pm 45/0/90]$ laminates has demonstrated that local delamination is responsible for fatigue failures at cyclic load levels below the static tensile strength. The circular symbols in the figure show the maximum cyclic load, P_{max} , plotted as a function of the number of load-controlled fatigue cycles, N , needed (1) to create delaminations along the edge in 0/90 interfaces (open symbols) and (2) to cause fatigue failures (solid symbols). The arrows extending to the right of data points at or near 10^6 cycles indicate runouts, i.e., no fatigue failures. The square symbols in figure 17 indicate the mean value of (1) load at onset of 0/90 interface delamination along the edge (open symbol) and (2) load at failure (closed symbol) in quasi-static tension tests. During these quasi-static tests, edge delaminations grew almost entirely through the specimen width before failure. Hence, the initial static tensile strength reflects the presence of large 0/90 interface edge delaminations. Yet the endurance limit for fatigue failure was 70 percent of this static tensile strength. Fatigue tests run at cyclic load levels below this limit, but above 40 percent of the tensile strength, contained extensive edge delaminations just like those observed in quasi-static tests. However, specimens in these tests did not fail in fatigue. But tests run at or above the 70-percent endurance limit also developed local delaminations in $+45/-45$ interfaces. These local delaminations, which originated from matrix cracks in the surface $+45^\circ$ plies, reduced the local cross section and changed the local stiffness. These changes in local stiffness and cross section increased the local strain in the 0° plies, resulting in fiber fracture and laminate failure. Future work will concentrate on predicting fatigue endurance limits using fracture mechanics models of local delamination.

- o O'Brien, T. K.: "The Effect of Delamination on the Tensile Strength of Unnotched, Quasi-Isotropic, Graphite/Epoxy Laminates," Proceedings of the SESA/JSME Joint Conference on Experimental Mechanics, Part I, Hawaii, May 1982, pp. 236.243.
- o O'Brien, T. K.: "Tension Fatigue Behavior of Quasi-Isotropic Graphite Epoxy Laminates," Proceedings of the 3rd Risø International Symposium on Metallurgy and Materials Science: Fatigue and Creep of Composite Materials, September 1982, pp. 259-264.
- o O'Brien, T. K.: "Analysis of Local Delamination and Their Influence on Composite Laminate Behavior," NASA TM 85728, January 1984 (Presented at the ASTM Symposium on "Delamination and Debonding of Materials," Pittsburgh, PA, November 1983).

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TENSILE FATIGUE BEHAVIOR OF UNNOTCHED $[\pm 45/0/90]_s$ GRAPHITE EPOXY LAMINATES



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FATIGUE OF NOTCHED COMPOSITE LAMINATES

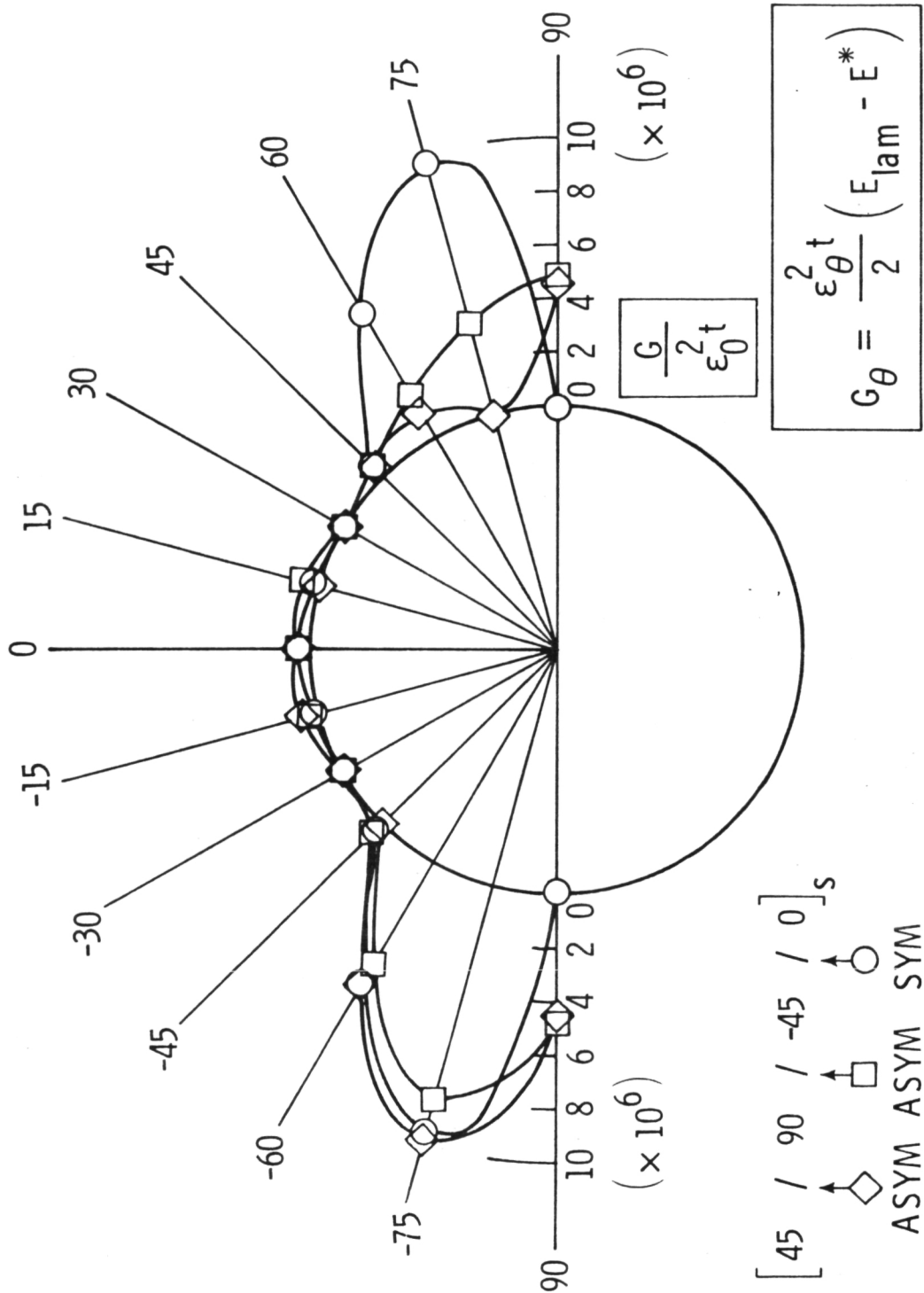
Project Engineer: T. Kevin O'Brien

The influence of matrix cracking and delamination on laminates with an open hole subjected to cyclic tension and/or compression loading is being studied. Some fatigue scan testing is being done in-house, as well as fracture mechanics analysis of delamination around an open hole. Under contract at VPI&SU (Stinchcomb), detailed damage studies are being conducted to determine such things as (1) the interaction between matrix cracking and delamination, (2) the influence of axial splitting on strain concentration near the notch, and (3) the influence of composite gage length on compression strength and life. A simple technique was developed in-house for calculating the strain energy release rate, G , for delamination around an open hole in a laminate. Discrete locations around the hole boundary were modeled as straight edges, with the ply orientations rotated by an appropriate angle. The circumferential strain, calculated from an elasticity solution, was substituted into a simple equation, derived from the rule of mixtures and laminated plate theory, to generate G distributions around the hole boundary. The G distributions shown in the figure were plotted for delamination in each unique interface of two quasi-isotropic laminates.

- o O'Brien, T. K., and Raju, I. S.: "Strain Energy Release Rate Analysis of Delamination Around an Open Hole in Composite Laminates," Proceedings of the 25th AIAA Structures, Dynamics, and Materials Conference, May 1984, Palm Springs, CA.

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STRAIN ENERGY RELEASE RATE DISTRIBUTIONS AT THE HOLE BOUNDARY FOR A QUASI-ISOTROPIC LAMINATE



COMPRESSION FAILURE MECHANISMS

Project Engineer: John D. Whitcomb

The particular mechanisms under study are free-edge delamination, microbuckling, and instability-related delamination growth.

Free-edge delamination in thin laminates has been studied extensively in recent years. But compression laminates are often thick. Of interest is whether thick laminates are more or less prone to delaminate than thin ones and where these delaminations are most likely to occur--i.e., between near surface plies or deep in the interior. An analytical study of thick laminates under uniaxial loads (ref. 1) showed that thick laminates should be less prone to delaminates than thin laminates, and when delaminations do occur, they should occur preferentially in the near-surface plies. The predictions agree with results reported by Charlie Harris and Don Morris from VPI&SU.

Fiber microbuckling is being studied for a wide range of resin stiffness, resin toughness, and laminate thickness. In the preliminary tests to date, microbuckling strains increased with resin stiffness. The effects of resin toughness and laminate thickness have not yet been identified. Related tests are being conducted by Tom Hahn at Washington University.

Instability-related delamination growth (IRDG) is growth driven by localized buckling of delaminated laminae. Although IRDG appears to be Mode I, geometrically nonlinear finite element analysis reveals that the growth is mixed mode. In fact, for long delamination lengths G_{II} governs IRDG (fig. 1). This work, has been published. It illustrates the need for careful stress analysis and for consideration of both Mode I and Mode II interlaminar fracture toughness. A systematic study of IRDG is just beginning under a grant to Delaware.

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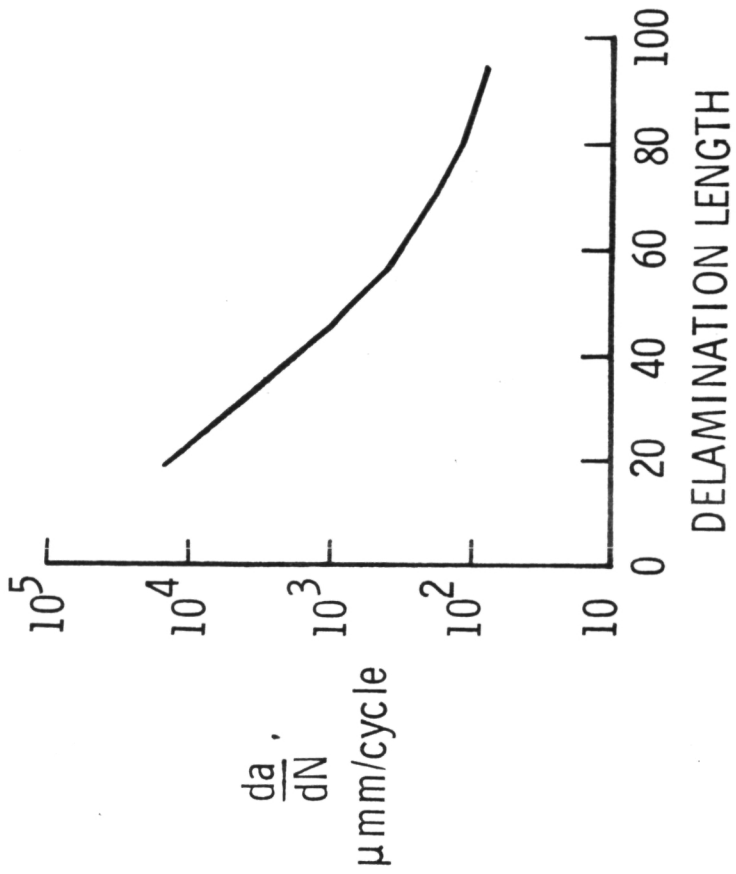
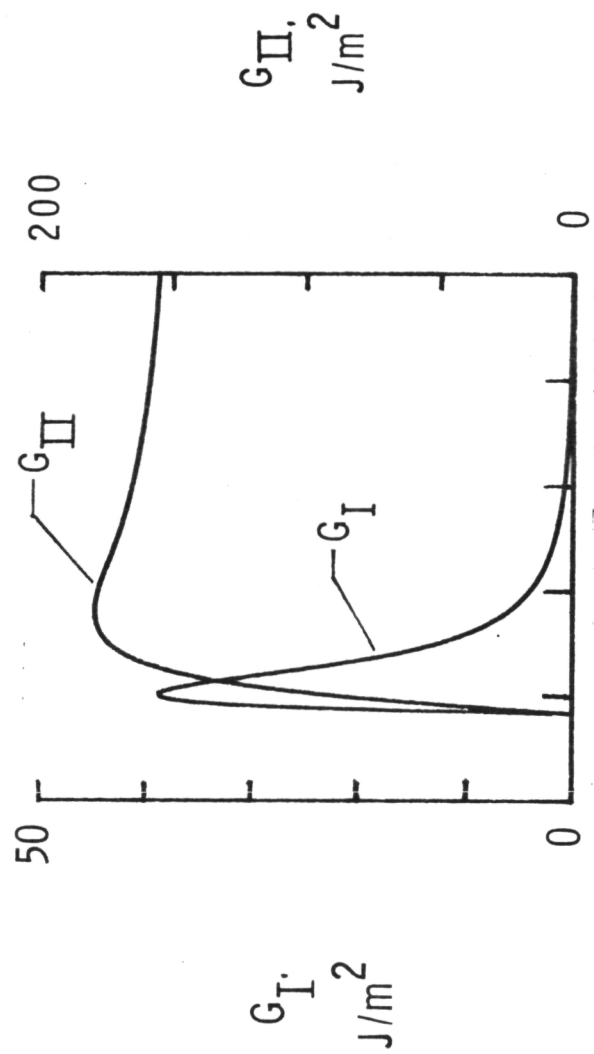
Whitcomb, J. D.; Raju, I. S.: Analysis of Interlaminar Stresses in Thick Composite Laminates With and Without Edge Delamination. NASA TM 85738, January 1984. Also presented at ASTM Symposium on Delamination and Debonding of Materials, Pittsburgh PA., November 1983.

Whitcomb, J. D.: Finite Element Analysis of Instability-Related Delamination Growth. Journal of Composite Materials, Vol. 15, September 1981, pp. 403-426.

Whitcomb, J. D.: Strain Energy Release Rate Analysis of Cyclic Delamination Growth in Compressively Loaded Composites. NASA TM 84598, January 1983. Also, presented at ASTM Symposium on the Effects of Defects in Composite Materials, San Francisco, CA., December 1982.

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G_I, G_{II} versus DELAMINATION GROWTH



- LARGE $\frac{da}{dN}$ ONLY WHEN G_I IS LARGE
- GROWTH EVEN WHEN $G_I \approx 0$
- G_{II} MUST BE DRIVING GROWTH

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STRESS ANALYSIS

Project Engineer: John Whitcomb

Recent efforts in the area of stress analysis have concentrated on developing efficient techniques for dealing with two types of problems: 1) thick laminates and 2) instability-related delamination growth.

The problem in analyzing thick laminates lies in accounting for all the individual plies. In particular, finite element analysis requires a sufficiently fine mesh to model both the geometrical shape and the variation in material properties from ply to ply. Numerous plies translate into large computer costs. One approach reported recently is to approximate the behavior of a thick laminate by that of two simpler laminates: a thin laminate which is representative of near-surface laminae and an infinitely thick laminate which is representative of interior laminae.

A more general technique for efficiently dealing with the variation in material properties from ply-to-ply is to include lamination effects within a single finite element. In conventional plane stress (or strain) or 3-D elements, lamination effects cannot be included, since only average extensional and shear moduli can be prescribed. The parameters are insensitive to stacking sequence. A new type of element has been developed which accounts for lamination effects within a single element. The rectangular laminated composite (RLC) element is currently implemented for plane stress or plane strain. However, the technique is sufficiently general to permit extension to quasi-3-D and 3-D analysis. This work is currently being documented and will be presented at the Society of Engineering Science meeting in Blacksburg in October.

The other area of stress analysis research is development of efficient techniques for analyzing instability related delamination growth. The required analysis is complicated by the inherent geometric nonlinearity of the mechanism. The figure illustrates one technique for simplifying the stress analysis. The original nonlinear configuration (a) is transformed into a linear configuration with two nonlinearly related loads, $(P_c - P_d)$ and $M(d)$. The linear configuration can be analyzed with linear finite elements and the loads can be determined using simple strength of materials. Using this transformed problem, closed form expressions can be derived which quantify the effect of various parameters on G_I and G_{II} . This technique is documented in reference 3.

Work to date on instability-related delamination growth has emphasized the response of a through-width delamination. Future emphasis will be on the three-dimensional counterpart, the embedded or patch delamination.

Shivakumar, K. N.; Whitcomb, J. D.: Buckling of a Sublaminate in a Quasi-Isotropic Composite Laminate. NASA TM 85755, February 1984.

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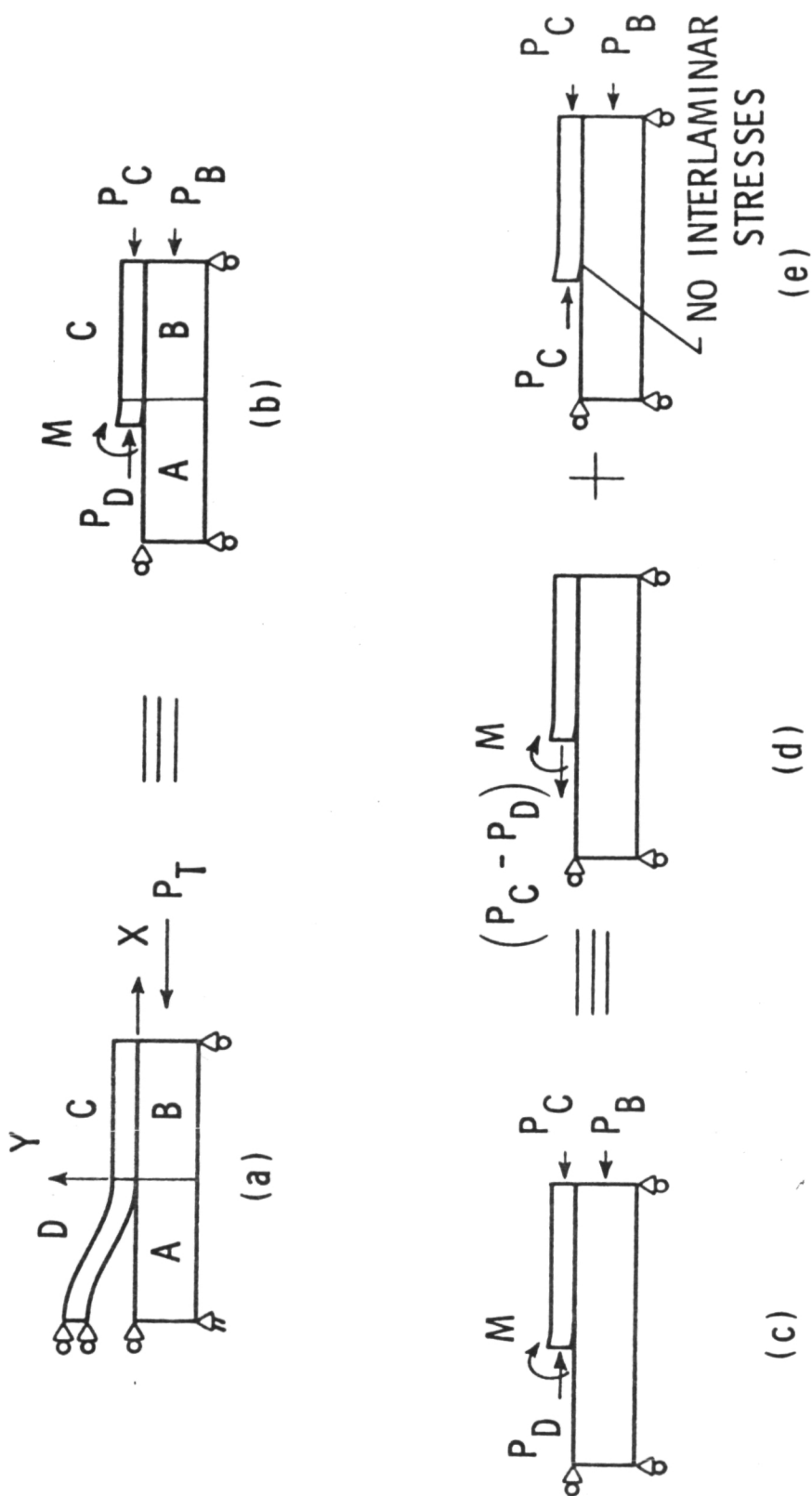


Fig. 2— Nonlinear configuration (a) transformed into linear configuration (d) with two nonlinearly related loads, $(P_C - P_D)$ and M .

MATERIALS CHARACTERIZATION

Project Engineer: John Whitcomb

Material properties related to fiber microbuckling and delamination growth are required. Some of the required data are available in the literature or from other activities in the branch related to tension behavior of composites. Data specifically generated for the compression study are dynamic G_{IC} data and static and fatigue mode I and combined mode I and mode II data. The dynamic G_{IC} data are being generated by Dr. Isaac Daniel of the Illinois Institute of Technology. The static and fatigue data were generated by Dr. R. Ramkumar of the Northrop Corporation. Only preliminary results are available from the dynamic G_{IC} study.

Based on the results generated at Northrop, delamination growth for a pure mode I situation is essentially a static problem. The extreme sensitivity of da/dn to changes in G_I makes it impractical to consider stable growth which could be characterized by the equation:

$$\frac{da}{dn} \approx CG^n$$

with $n \approx 5 - 6$

Some specimens had compressive stresses normal to the delamination. The compressive stress appeared to retard the mode II delamination growth.

MICROMECHANICS OF INTERLAMINAR FRACTURE

Project Engineer: John H. Crews

A NASA sponsored workshop in 1982 identified the need for an accelerated program in micromechanics as an aid in determining a path towards tougher composites. We have structured a two-part program for funding made available in FY 84 - FY 85. The first part will attempt to establish analytically which bulk resin test most closely predicts the interlaminar toughness of a composite made from that resin. The second part will establish experimentally which predominant deformation mechanism (crazing, shear banding, etc.) consistently provides the matrix materials with the highest interlaminar toughness.

In the analytical studies, three-dimensional stress analyses of a delamination front in a composite will be generated using finite-element methods. Similarly, a variety of neat-resin fracture specimens will be analyzed to determine stress states and strain energy release rates.

The results from composite delamination tests and neat-resin fracture tests can then be experimentally compared to determine which parameters control the fracture and which neat-resin specimens most closely predict the delamination.

In determining which deformation mechanisms provide matrix materials with the highest interlaminar fracture toughness, a number of matrices will be selected each displaying one dominant mode of inelastic deformation, such as shear banding. The matrices will be evaluated both in neat-resin fracture and in composite delamination tests. The occurrence of their primary deformation modes in the composites will be verified by microscopy.

If as a result a particular deformation mechanism consistently produces the highest delamination toughness, these results will be beneficial to polymer chemists searching for means to further toughen these materials.

The analytical work is already in progress through the ongoing work of Adams at Wyoming, and Raju at Langley. Preliminary experimental programs in this study are being carried out by Bradley at Texas A&M. Contract negotiations for additional contributing efforts are in progress.

LAMINATE BEARING RESPONSE

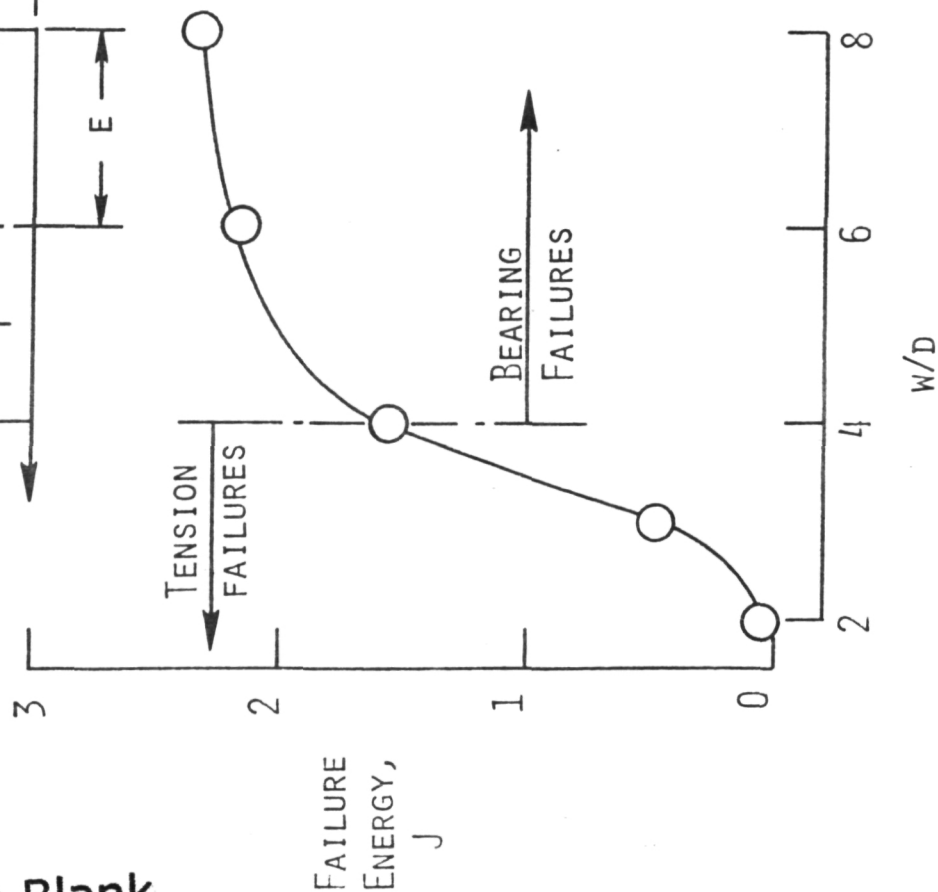
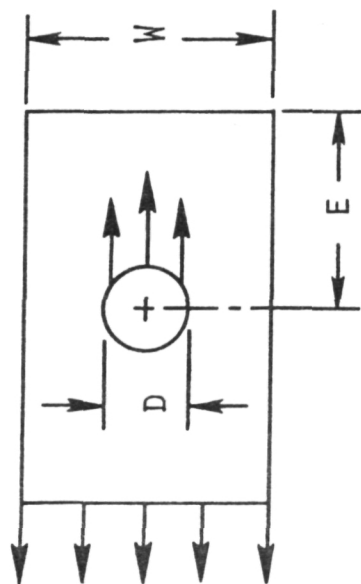
Project Engineer: John H. Crews, Jr.

Bolted composite joints are typically designed for ultimate failures to occur in the net-tension mode because current composites have their highest strength in this failure mode. Unfortunately, however, net-tension failures propagate in a brittle catastrophic manner, in contrast to bearing failures which are usually stable and dissipate much more energy. For these reasons, the bearing mode would be preferred if composites with higher bearing strengths could be developed. A program has been developed to investigate the basic laminate response for net-tension and bearing failures. This study should indicate ways to improve the toughness for net-tension failures and to improve the strength for bearing failures. The current program consists of two parts. The first part analyzed laminates subjected to tension-reacted-bearing (TRB) loads. The second part involves laminate failure analyses for combined bearing and bypass loading in tension as well as compression.

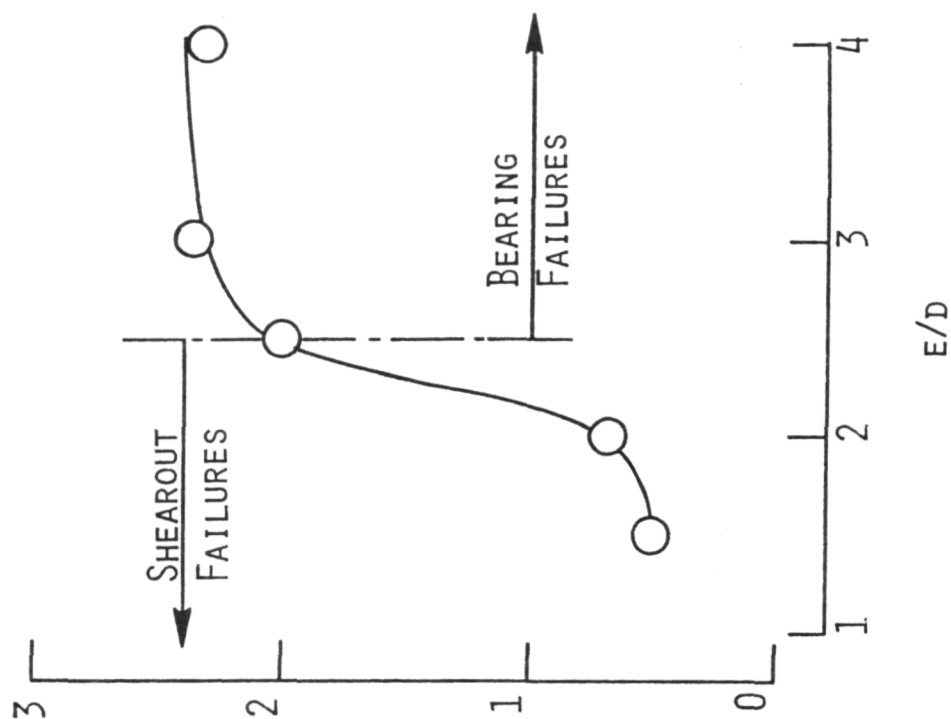
In the TRB study, a finite element analysis has been conducted to calculate stresses around a fastener hole for ranges of specimen configurations. Then, specimens were designed to fail by either the net-tension mode, the bearing mode, or the shearout mode. The damage onset and ultimate strength have been measured for each such mode. The local damage and the corresponding ply stresses are currently being used to analyze the three failure modes. Failure energies were measured. The attached figure shows Thornel 300/Narmco 5208 (T300/5208) laminates are toughest when failed by the bearing mode, as expected. Results from this study will be presented at an ASTM meeting in the fall.

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HOLE FAILURE ENERGY



T300/5208
16-PLY QUASI-ISOTROPIC
 $D = 6.35 \text{ MM}$



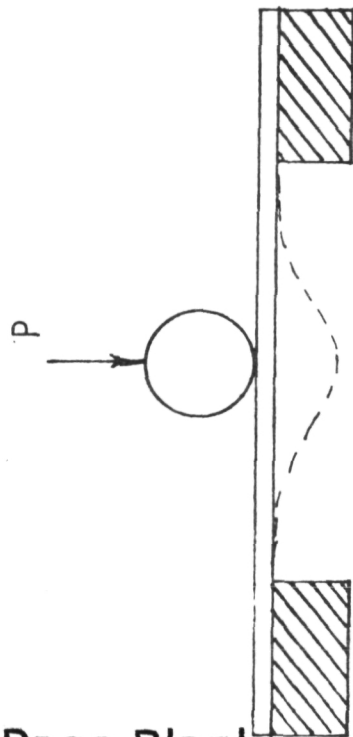
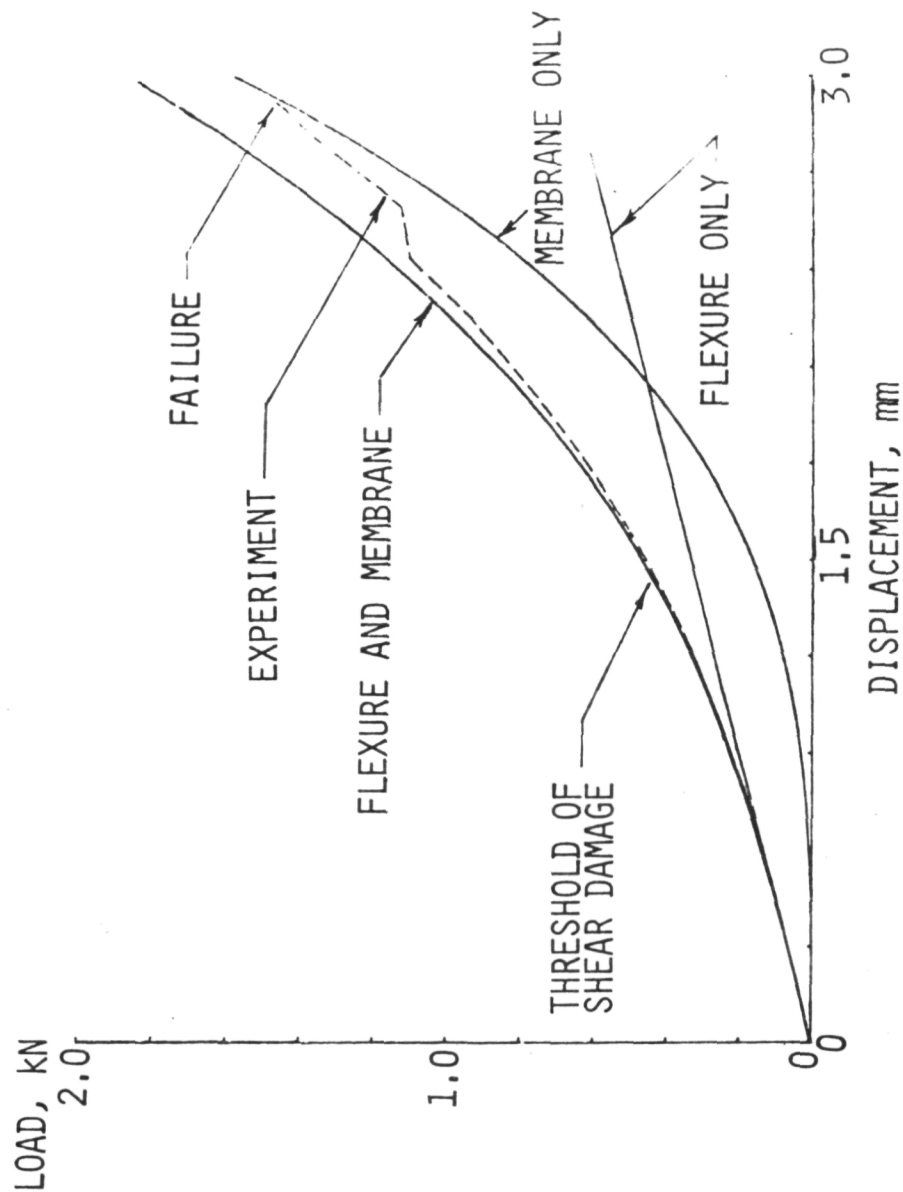
STATIC INDENTATION TESTING TO DETERMINE IMPACT RESISTANCE OF COMPOSITES

Project Engineer: Gretchen Bostaph

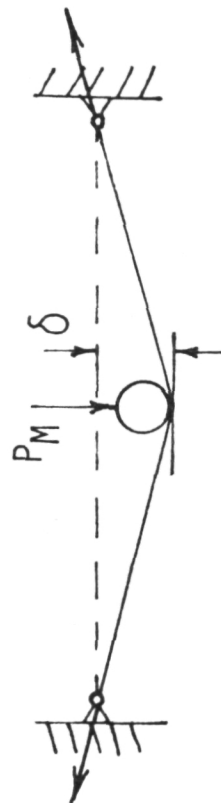
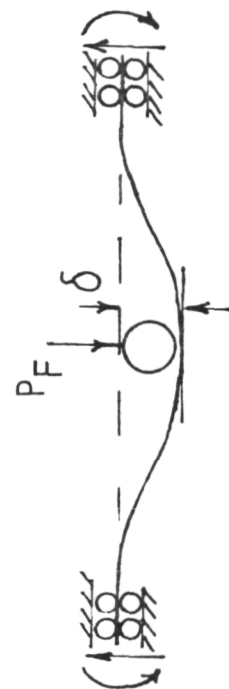
The objective of our basic research studies is to develop a good understanding of, and the ability to mathematically model, the impact deformation and damage mechanisms. The effects of such damage on load-carrying ability are being investigated in industry, and by Jerry Williams in the Structures Division. Lee Gause, of the NADC, has shown that low velocity impacts on thin composite plates can be accurately simulated by means of an equivalent static indentation test. Static tests were therefore used in this study because of their simplicity. However, while it has been shown that static tests and impact tests agree with respect to deformation mechanics, the damage sequences may still differ. Sierakowski, for example, found the formation of generator strips starting from the top ply under impact. Our static tests on the same material showed the development of such strips only later during the penetration phase, after the failure of the lower surface plies. The first study was restricted to establishing the significance of the membrane contribution to the deformation behavior. Our results have shown that even undamaged laminates deflect so much under impact that any analysis must account for membrane effects. An analysis has been developed in which the initial plate deformation has contributions from flexural, membrane, and indentation forces. As shear delaminations progress through the plate, the flexural resistance weakened until the plate behaves virtually as a pure membrane. The final failure occurs when the membrane stresses exceed the fiber failure strength. The figure shows the calculated load-displacement curves for an undamaged plate and for a fully delaminated plate. Actual plate behavior should be bounded by these two extreme cases. The analysis correctly predicts the initial plate response, and the figure shows how the data curve approaches the membrane curve. For all test cases, the results showed that the analysis correctly predicts boundaries for the plate deformation behavior.

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$$\text{FLEXURAL REACTION} + \text{MEMBRANE REACTION} = \text{TOTAL REACTION}$$

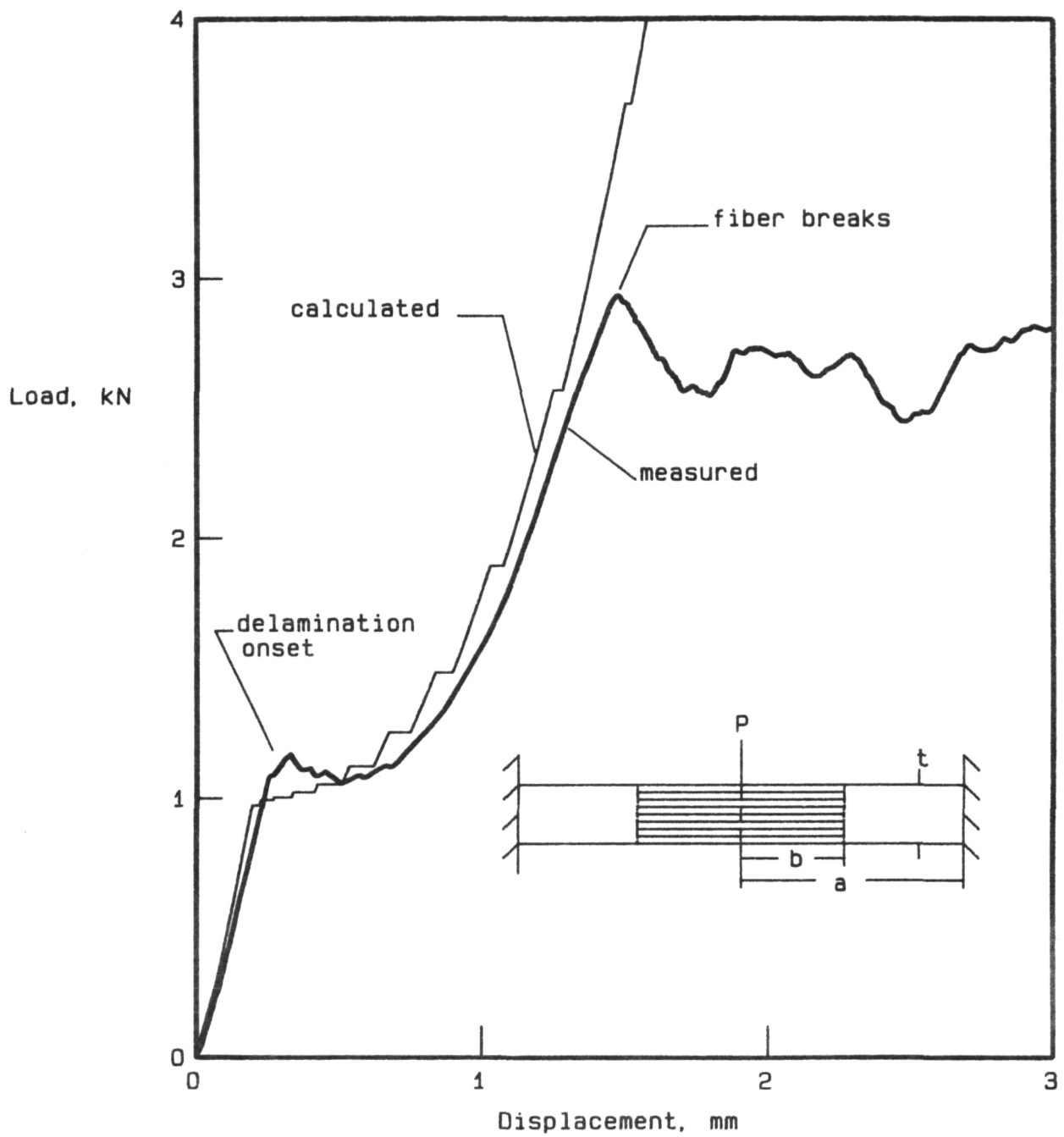


A FRACTURE MECHANICS APPROACH TO IMPACT DELAMINATION DAMAGE

Project Engineer: Gretchen Bostaph

The objective of this second study is to be able to relate impact damage threshold and delamination progression in composites to basic fracture mechanics properties which can be used to screen matrix materials. The earlier analysis was able to model the initial elastic behavior of statically loaded circular composite plates, and also to predict the plate behavior after complete delamination. The analysis could not, however, predict the delamination threshold or the damage progression. A new plate solution uses fracture mechanics properties to model the delamination progression. The initial damage threshold is controlled by matrix shear strength, whereas the delamination growth criterion is the material strain energy release rate, G_{IIC} . From this analysis, artificial load-deformation functions were generated and were compared to static indentation test results for circular plates of a range of thicknesses and diameters. The test data show that the analysis accurately predicts the threshold damage point and damage progression for various thicknesses and support conditions. However, the dominant failure modes can differ. Thin, or large diameter plates tend to be more membrane critical, while thicker, or smaller diameter plates are more shear and flexure critical. The model indicates that damage and shear strain energy release rate are closely related. It also predicts the influence of improving the matrix toughness. This work will be continued with modified techniques to describe the damage progression in thicker plates impacted by sharper indentors.

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EFFECTS OF MATRIX ON COMPOSITE IMPACT RESISTANCE

Project Engineer: Gretchen Bostaph

The objective is to determine the effect of matrix properties on the impact susceptibility of thin composite plates.

Several different tough matrix materials will be tested quasi-statically to determine their load-displacement behavior. The test data will be compared to an existing fracture mechanics analysis which uses basic material properties to predict delamination damage threshold and damage progression. The criteria for initial damage and damage growth are the matrix shear strength and shear strain energy release rate. The damage threshold is calculated based on a bulk stress criterion in the matrix. Since the greatest shear stresses occur over the cylindrical indentation area under the point load, this shear stress determines the damage initiation load. Previous studies have shown good correlation between expected initiation loads and experimentally determined loads for a range of thicknesses of a brittle matrix material, 5208/T300. The results confirm that damage initiation is matrix shear strength controlled and improvements in impact susceptibility should be attainable by using tougher matrix materials.

No results yet.

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IMPACT RESISTANCE OF THICK FILAMENT-WOUND CYLINDERS

Project Engineer: W. Illg

Filament winding is a cheap procedure to fabricate cylindrical pressure vessels of continuous-fiber composites. The resulting vessels are also lighter than their metal counterparts. Consequently, NASA is replacing the steel solid-fueled booster rocket casings on the Shuttle spacecraft with lighter graphite/epoxy filament-wound cylinders (FWC). These will be large structures--12 feet in diameter, 1.3 to 1.6 inches thick, and 20 feet long. Thus, transportation and handling are major operations that are subject to accidental damage. Of concern is the effect of such damage. In support of Marshall Space Flight Center, the responsible office for FWC boosters, we are studying the effect of impact damage on the residual tensile strength of graphite/epoxy FWC.

In this study, we will focus on the impact energies that produce barely visible damage. Damage will be inflicted by various shaped indenters to simulate many possible accidental encounters. The extent of damage will be measured ultrasonically and radiographically before tensile tests. The damage measurements will be correlated with strength reductions via fracture mechanics analogs. This study must supply quantitative information in a short time to contribute to the Shuttle FWC project.

At the same time, this project activity will focus and combine our existing generic research in impact and fracture.

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EFFECT OF PARTIAL BONDING ON IMPACT RESISTANCE

Project Engineer: W. Illg

When a composite is impacted, both fiber and matrix damage can occur. Under compression load, such damage can lead to buckling and shear crippling. High matrix strength and stiffness are required to suppress these failure modes. Under tension loading, failure is initiated when a crack can propagate unstably. C. C. Poe has shown that large damage areas at crack tips effectively blunt the crack and increase the toughness. In past work, Professor Atkins has shown that intermittently polyurethane-coated boron fibers displayed a higher fracture toughness than clean, well-bonded fibers. Later, both Elber and Felbeck studied the effect of partial interlaminar bonding on toughness and found significant improvements using perforated mylar between laminae.

In this study we examined this concept parametrically and investigated its influence on compressive strength at the same time. While the partial bonding may show a marginal improvement for plates tested in tension, the results in the figure show clearly that the effective reduction in matrix toughness destroys the impact resistance of the graphite/epoxy plates under compression. The work will be continued to evaluate the partial bonding concept in thicker laminates. Both in our impact work and in the buffer strip work, the concept represents a study method rather than a useful toughening method for aerospace structures.

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IMPACT UNDER COMPRESSION

24 PLIES T300/5208

Partially Bonded

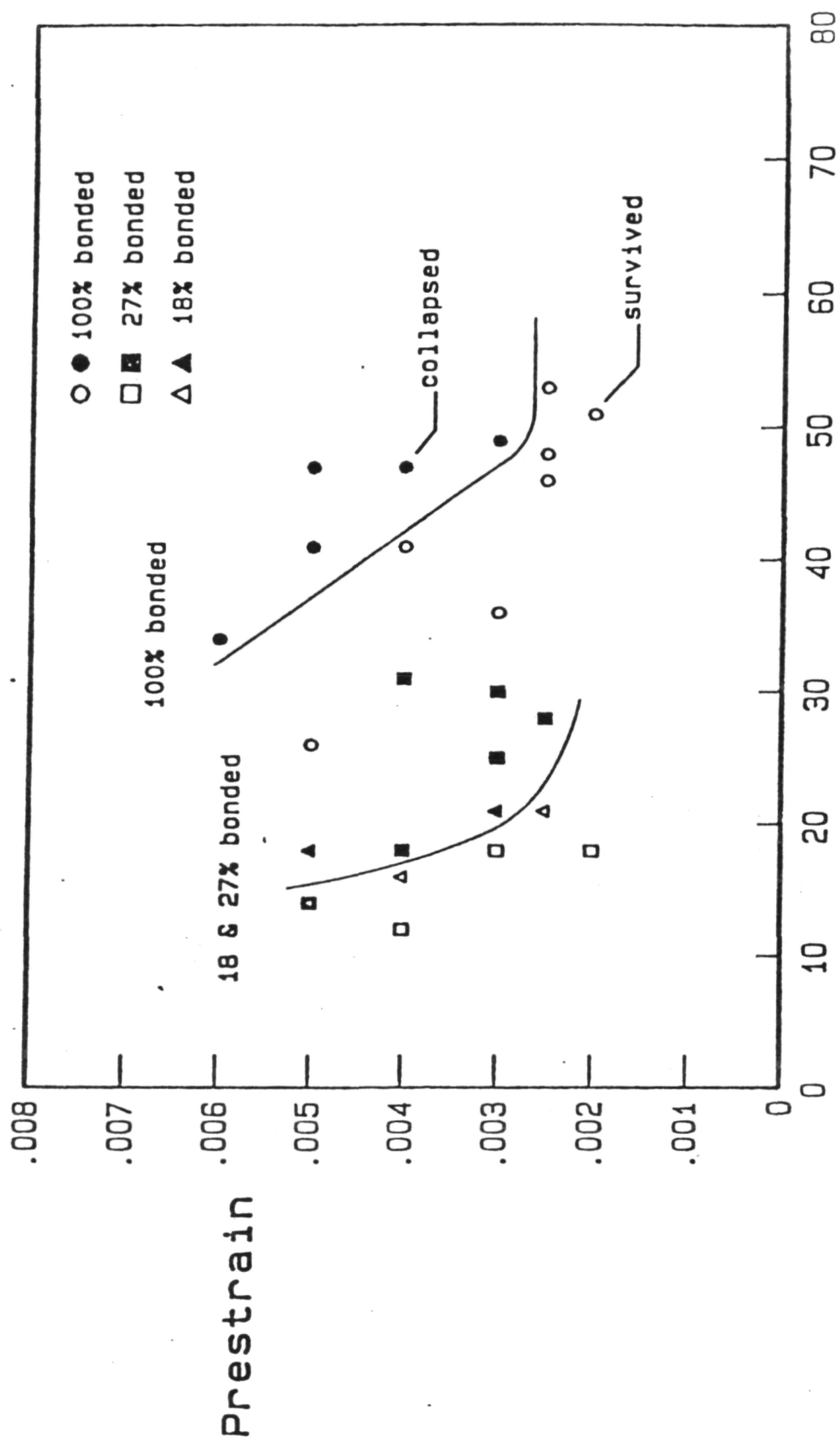


Figure 2

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RELATED IMPACT STUDIES UNDER GRANTS

Project Engineer: W. Illg

Two aspects of the impact event which profoundly affect the accumulation of damage are the details of the contact behavior near the indenter and the magnitude of internal transient strains. Two grants treat these topics. At Purdue University, C. T. Sun is extending his earlier studies of contact behavior to include the effects of preloads. For a [0₅/90₅/0₅] layup, he found that tension preloads increase depth of indentation, whereas the opposite occurs for compression preloads. In his tests, he found that the area of delamination damage correlated strongly with impact velocity. This work will continue to look at the equivalence of static and dynamic loading and the sequence of damage accumulation.

The magnitude of internal transient strains is being measured in graphite/epoxy specimens by I. M. Daniel at the Illinois Institute of Technology using the embedded gage technique, which he pioneered. Most of the effort thus far has concentrated on refining the embedment procedure for graphite/epoxy.

To help develop a predictive methodology for impact damage, K. N. Shivakumar of Old Dominion University is developing an energy-based model to calculate damage. The model uses energy principles to determine impact-force history, and then splitting and fiber breakage threshold levels are determined using strength criteria. This model will be refined to correlate more closely with experimentally determined damage accumulation.

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FRACTURE OF COMPOSITES

Project Engineer: C. C. Poe, Jr.

Unstable fracture will be an important part of damage tolerant design for composite primary structures. The many possible layups and materials make selecting fracture resistant composites more complex than for metals. In a two-year program, a single fracture toughness parameter was developed for all continuous fibrous composite laminates that can be used to predict fracture toughness from only lamina properties. The predictions are valid except when large splits develop at the crack tips. Then toughness is underpredicted. The figure shows predicted toughness versus ultimate tensile strength for T300/epoxy $[0_m/45_n/-45_n/90_p]_S$ laminates. Toughness is bounded above by $[0_m/45_n/-45_n]_S$ laminates and below by $[0_m/90_p]_S$ laminates. Notice that, for laminates with equal

Young's moduli, toughness is a constant even though ultimate tensile strength varies greatly. Thus, strength retention with damage or notch insensitivity (ratio of toughness to ultimate tensile strength) varies greatly with layup. For a given proportion of 0° plies, $[0_m/90_p]_S$ laminates are the most notch sensitive, $[0_m/45_n/-45_n]_S$ laminates are the least, and notch sensitivity increases with an increase in proportion of 0° plies. However, the increase in notch sensitivity of $[0_m/90_p]_S$ laminates with an increase in proportion of 0° plies is very small. Crack-tip splitting, which tends to occur most in laminates with a large proportion of 0° plies and a small proportion of 45° plies, can greatly increase toughness and reduce notch sensitivity.

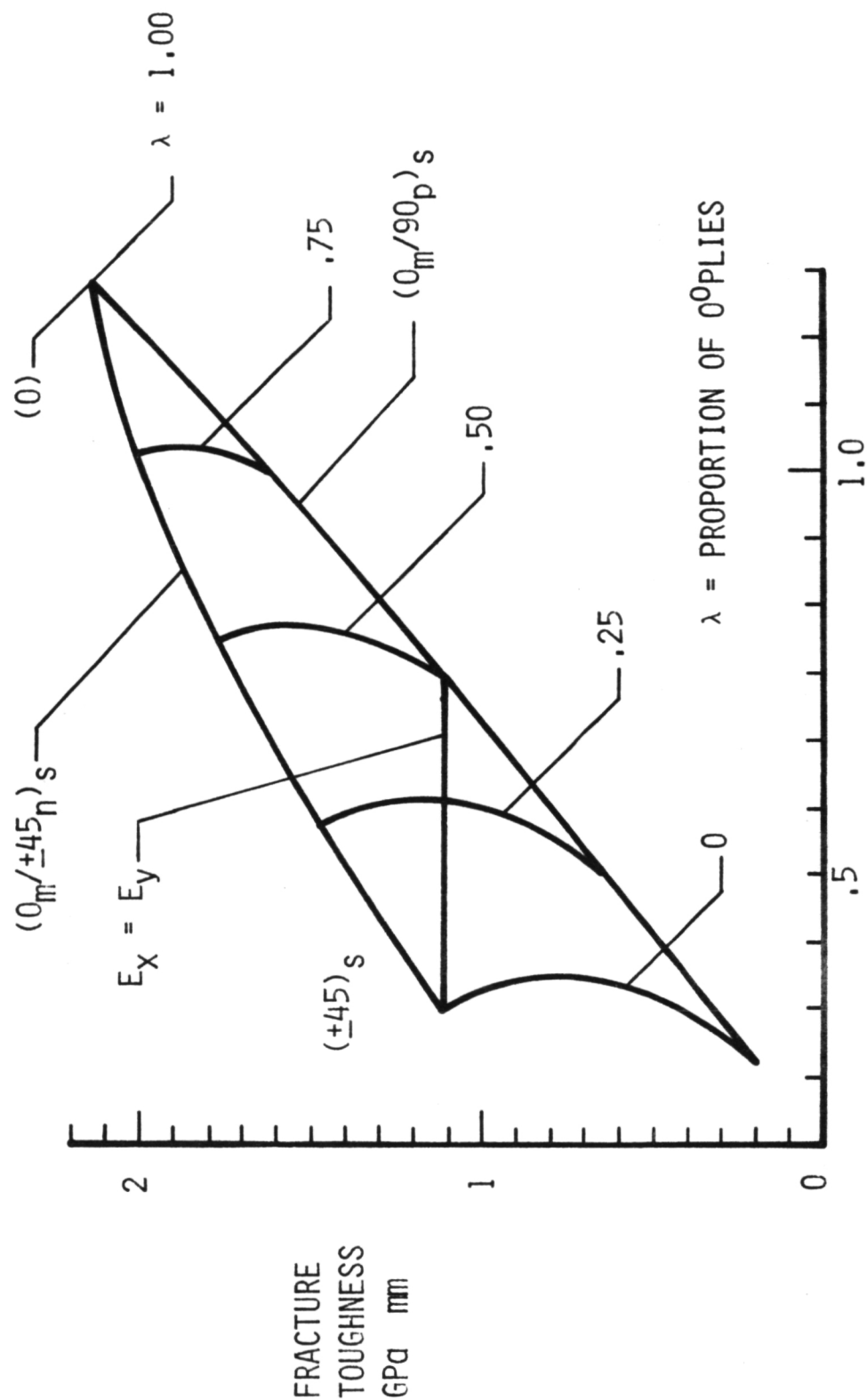
Studies also indicate that toughness increases with fiber strength and not fiber failing strain (unless, of course, Young's modulus of the fiber is constant), and that the toughness of hybrid laminates with stiffer off-axis plies is greater than that of laminates made with only one fiber. Matrix stiffness has little effect on toughness, but matrix strength greatly reduces toughness when crack-tip splits tend to develop. Therefore, strong, tough matrices that preclude splitting will give the minimum toughness values in the figure.

Studies on thick laminates (up to 120 plies) indicate that toughness decreases with thickness. Crack-tip damage develops only in the surface plies and is much smaller in extent than that in thin laminates. Thus, the toughness of thick laminates tends to be a minimum, like the predicted values in the figure. Tests of thick laminates with elliptical surface flaws indicate that toughness is the same for surface cracks and through-the-thickness cracks, much like metals.

In the future, studies of surface flaw behavior will be continued and studies of fracture under combined bending and axial loads will be initiated.

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FRACTURE OF COMPOSITES $(0_m/+45_n/90_p)_s$ T300/EPOXY



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FRACTURE OF THICK COMPOSITE LAMINATES

Project Engineer: C. C. Poe, Jr.

Don Morris of VPI&SU (Grant NAG-1-343) is experimentally determining the influence of thickness on fracture toughness of composite laminates. Several layups and types of specimens as thick as 120 plies are being investigated. The specimens contained through-the-thickness crack-like flaws. For the most part, toughness decreases with thickness, much like metals, and is the same as measured with center-cracked, three-point-bend, and compact specimens. Crack-tip splitting and delaminating elevates toughness by reducing local stresses. The investigation discovered that crack-tip damage decreases in extent with thickness, and damage develops only in the plies near the surface. Thus, crack-tip damage does not elevate the toughness of thick laminates like thin laminates. Laminates of intermediate thickness behave somewhere in between.

Tests are also being conducted on thick specimens with part-through crack-like flaws of elliptical shape. The laminates are quasi-isotropic. Preliminary results indicate that the toughness with through-the-thickness and part-through flaws is equal.

In the future, the investigation will look at fracture under a combination of bending and tension. Until now, all laminates have been made with prepreg tape material. Future work will also include filament-wound laminates.

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DAMAGE TOLERANCE CONCEPTS

Project Engineer: C. C. Poe, Jr.

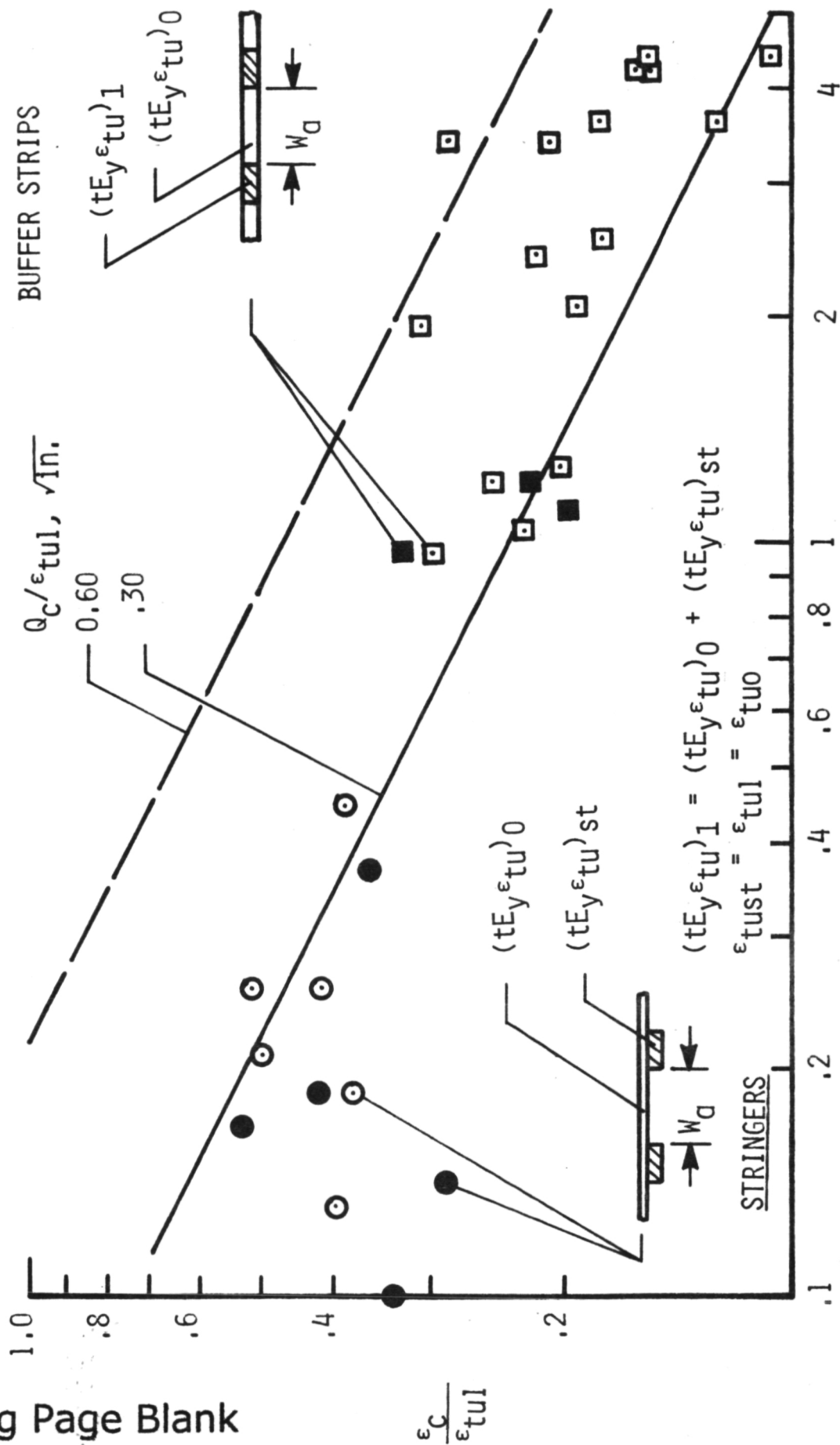
Studies were made to determine how much damage tolerance stringers and buffer strips provide in composites. Panels were made with T300/5208 material with both $[45/0/-45/90]_{2S}$ and $[45/0/-45/0]_{2S}$ layups. The stringer panels were 12 inches wide. The stringers were unidirectional and co-cured with the sheet. They were made with various thicknesses and widths. The buffer strip panels had widths from 10 to 35 inches. The buffer strips were made with both S-glass and Kevlar-49 tape material, and with various widths and thicknesses. The panels were cut in the center to simulate damage and were loaded to failure in tension.

Results are shown in the figure. Cracks were arrested in both types of panels, and failure strains were much larger than plain sheets. A fracture mechanics type of analysis that predicts the effects of spacing, thickness, and material of the buffer strips and stringers was developed for both types of panels. The failing strains are plotted in the figure against stringer or buffer strip spacing, which is the arrested crack length, times a factor that accounts for the thickness and material of the buffer strips and stringers and a factor "xi" that accounts for the sheet layup. The failing strain of the panel is normalized by the failing strain of the buffer strips or, in the case of the stringer panels, the failing strain of the sheet. The failing strain of the S-glass buffer strips is 2.92 times that of the sheet, and the failing strain of the Kevlar-49 buffer strips is 1.68 times that of the sheet. Logarithms are plotted for convenience. The upper curve is predicted like the lower curve, but with twice the toughness. Because of extensive crack-tip damage, which is not taken into account, many of the buffer strip panels failed near the upper curve, that is, at twice the predicted strength. Stringer panels with thick stringers failed below the predicted curve because bending effects reduced the effectiveness of the stringers. These effects can be taken into account empirically.

Future work will attempt to predict the effects of crack-tip damage and stringer bending.

DAMAGE TOLERANCE CONCEPTS

OPEN SYMBOLS - (45/C/-45/90)_{2S}, FILLED SYMBOLS - (45/0/-45/0)_{2S}



$$w_d \epsilon^2 (tE_y \epsilon_{tu}^2)_0 / (tE_y \epsilon_{tu}^2)_1, \text{ IN. WHERE } \epsilon = 1 - \nu_{yx} \sqrt{E_x/E_y}$$

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SHEAR-LAG ANALYSIS OF COMPOSITES

Project Engineer: C. C. Poe, Jr.

Jim Goree of Clemson University (Grant NSG-1297) is developing a shear-lag analysis that models splitting and delaminating at the tips of a crack-like flaw. The fibers in the loading direction are assumed to carry all of the normal force and matrix is assumed to carry all of the shear force. The off-axis plies are assumed to act as shear springs that act in parallel to the matrix. Strengths have been successfully predicted for unidirectional boron/aluminum. For laminates that split at the crack tips, some in-house work with this model has successfully predicted how the fracture toughness, which is elevated by splitting, increases with crack length.

Future work will attempt to apply the shear-lag analysis to fracture of buffer strip panels that develop significant crack-tip damage in the buffer strips.

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EFFECT OF DEBOND GROWTH ON CRACK PROPAGATION IN COMPOSITE PLATES
REINFORCED WITH ADHESIVELY BONDED COMPOSITE STRINGERS

Project Engineer: C. A. Bigelow

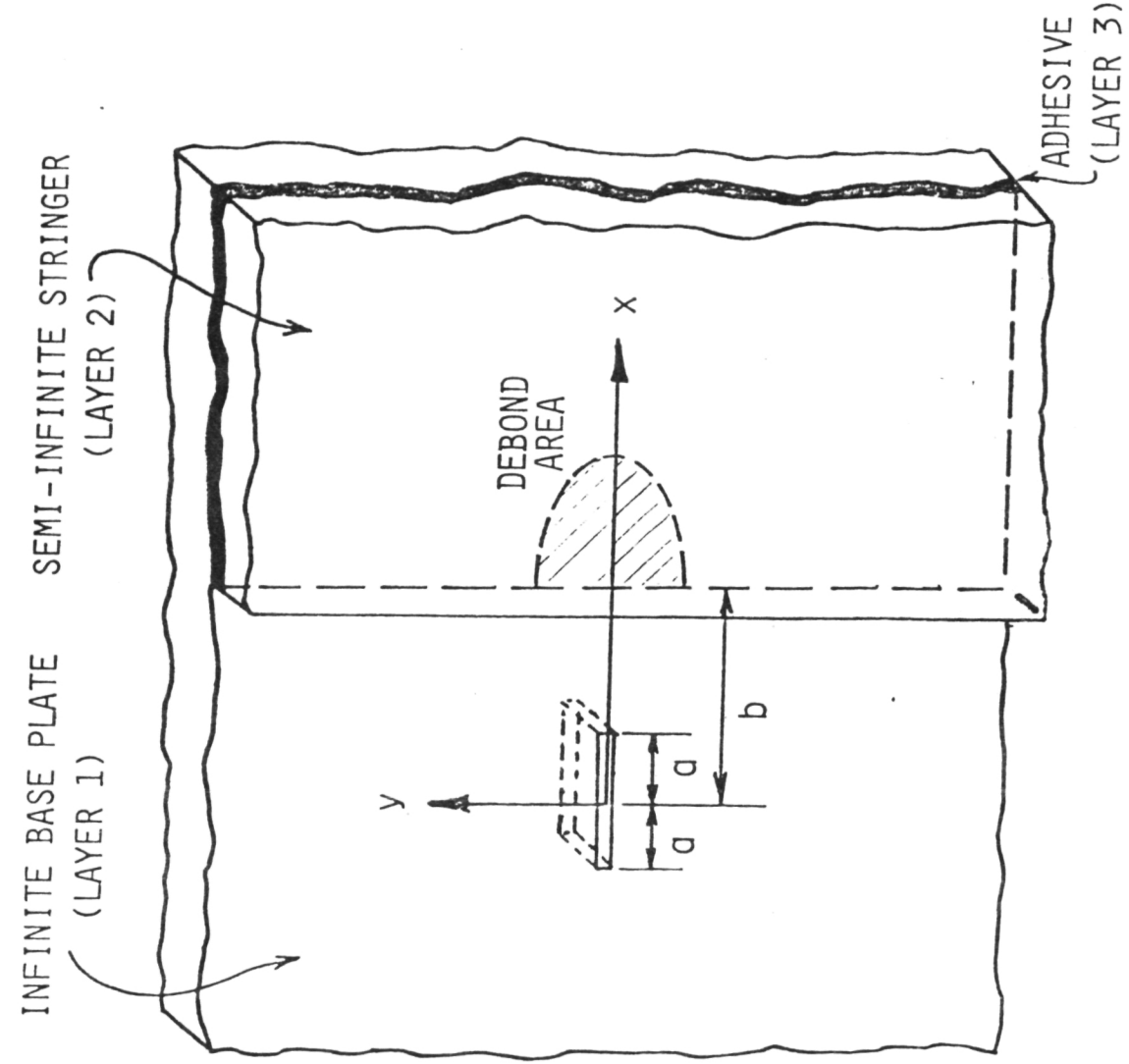
Like buffer strips, stringers are widely used as stiffening elements to decrease the possibility of fracture initiation or to arrest crack propagation. Many authors, including Arin, Hart-Smith, and Swift, have investigated the interaction of a crack with a stringer. Experimental work has shown that as the crack tip approaches the stringer, debonding may start and propagate through the adhesive. The objective of this study will be to develop an analysis to predict the effect of debond growth on crack propagation in composite plates reinforced by adhesively bonded stringers. Using Green's functions, an integral equation describing the problem will be sought. The solution will be formulated using the complex variable elasticity theory developed by Muskhelishvili and by Lekhnitskii.

In the stringer configuration shown in the figure, it is assumed that the debond area and crack length are small compared to the width of the stringer. Based on this assumption, the stringer will be modeled as a semi-infinite plate bonded to an infinite base plate, with a finite crack in the base plate. Both layers are made of a fiber-reinforced composite which will be treated as a homogeneous, orthotropic medium. The stringer and the base plate are bonded together within an adhesive layer of constant thickness. Debonding may occur within the adhesive layer. The problem to be solved is the following: For a given applied stress state T_y acting on the plate away from the crack, a given crack length $2a$, and a given adhesive fracture strength in shear τ_p , find the stress-intensity factor at the crack tip and determine the boundary of the debond area. The problem will be solved under the assumptions that all parts are homogeneous and linearly elastic, and the plate and stringer are thin, orthotropic layers in plane stress, with the adhesive shear transfer considered as a body force without eccentricity. The adhesive layer will be modeled as a shear spring.

Using the resulting continuity equations, it is possible to write a system of integral equations for the unknown interlaminar shearing stresses τ_x and τ_y . The integral equations will be solved numerically by reducing them to a system of algebraic equations. Once the stress distribution in the adhesive, $\tau_x(x,y)$ and $\tau_y(x,y)$, is known, the stress-intensity factors at the crack tips can be determined.

When the stress distribution of the composite structure is known, it will be possible to predict the initiation and extent of debonding in the adhesive layer. The problem can be solved by introducing an adhesive failure criterion. Then the boundary of the adhesive debond region may be determined concurrently with solving the integral equations. This analysis is directed toward my doctoral dissertation.

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PROBLEM CONFIGURATION

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FATIGUE OF GRAPHITE/EPOXY BUFFER STRIP PANELS

Project Engineer: C. A. Bigelow

Some applications of graphite/epoxy structures will undoubtedly demand higher levels of damage tolerance than the intrinsic properties of fiber and matrix can supply. The concept of buffer strips which periodically interrupt possible crack paths by either forming intense delamination zones or by placing a low modulus, high strain capability fiber bundle in the way of the crack has been successfully evaluated before, for example, the experimental program by Poe and Kennedy.

In this project, buffer strip toughened panels are tested under fatigue loading of elevated temperature and humidity. So far in the program, precracked buffer strip panels have been tested under spectrum loading at room temperature to establish a baseline data set.

The graphite/epoxy buffer strip panels were 101.6 mm wide, with S-glass or Kevlar-49 buffer strips 5.08 mm wide spaced 25.4 mm on center. A slit 10.16 mm wide was machined in the center of each panel to represent damage. The basic layup was $[45/0/-45/90]_{2S}$, with four plies of buffer material.

Each panel was subjected to a prerecorded spectrum loading. Two levels of maximum strain and three different lifetimes were used. One group of specimens was fatigued using the spectrum loading, then statically loaded in tension to determine the residual strength. The other group of specimens was first statically loaded in tension until the crack grew into the buffer strip. The spectrum loading was then used on the specimen. After fatiguing, each specimen was statically loaded in tension until failure to determine the residual strength.

During the static loading of each specimen, load, crack-opening displacement (COD), remote longitudinal strain, and strain in the buffer strip at the crack tip were plotted. At selected intervals during the fatigue cycle, the loading was stopped and the specimen was statically loaded to the maximum allowable strain used in the fatigue cycle. The specimen was also radiographed if there had been any change in the slope of the load versus COD curve. A change in the slope of this curve could indicate crack extension.

At the strain levels used, the fatigue cycling had no discernible effect on the residual strengths. Preloading the buffer strip panels statically before the fatigue loading also did not affect the residual strengths.

The next step will be testing the buffer strip panels at elevated temperature and with moisture conditioning.

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BUFFER STRIPS IN COMPOSITES AT ELEVATED TEMPERATURE

Project Engineer: C. A. Bigelow

Fibrous composite materials like graphite/polyimide are light, stiff, and strong. They have great potential for reducing weight in aircraft structures. However, fibrous composites are usually notch sensitive and, therefore, lose much of their strength when damaged. Poe and Kennedy have shown that buffer strips greatly improve the damage tolerance of graphite/epoxy laminates loaded in tension. These narrow, parallel buffer strips are made into the laminate itself by interrupting and replacing plies of the laminate with another material or layup. These buffer strips can arrest fracture and increase the residual strength of the damaged laminate. However, the buffer strip concept has been evaluated only at room temperature.

The present investigation evaluated the buffer strip concept at elevated temperature. Accordingly, graphite/polyimide buffer strip panels were tested at room and elevated (177°C) temperatures. One layup was used: $[45/0/-45/90]_{2s}$. The buffer strip material was 0° S-glass/polyimide. The buffer strips were made by replacing narrow strips of the 0° graphite plies with strips of the 0° S-glass on either a one-for-one or a two-for-one basis. All panels had the same buffer strip spacing and width. Slits were cut in the center of each panel to represent damage. Each panel was loaded in tension until it failed. During the tests, panels were radiographed and crack-opening displacements were recorded to indicate fracture onset, fracture arrest, and the extent of damage in the buffer strip after crack arrest.

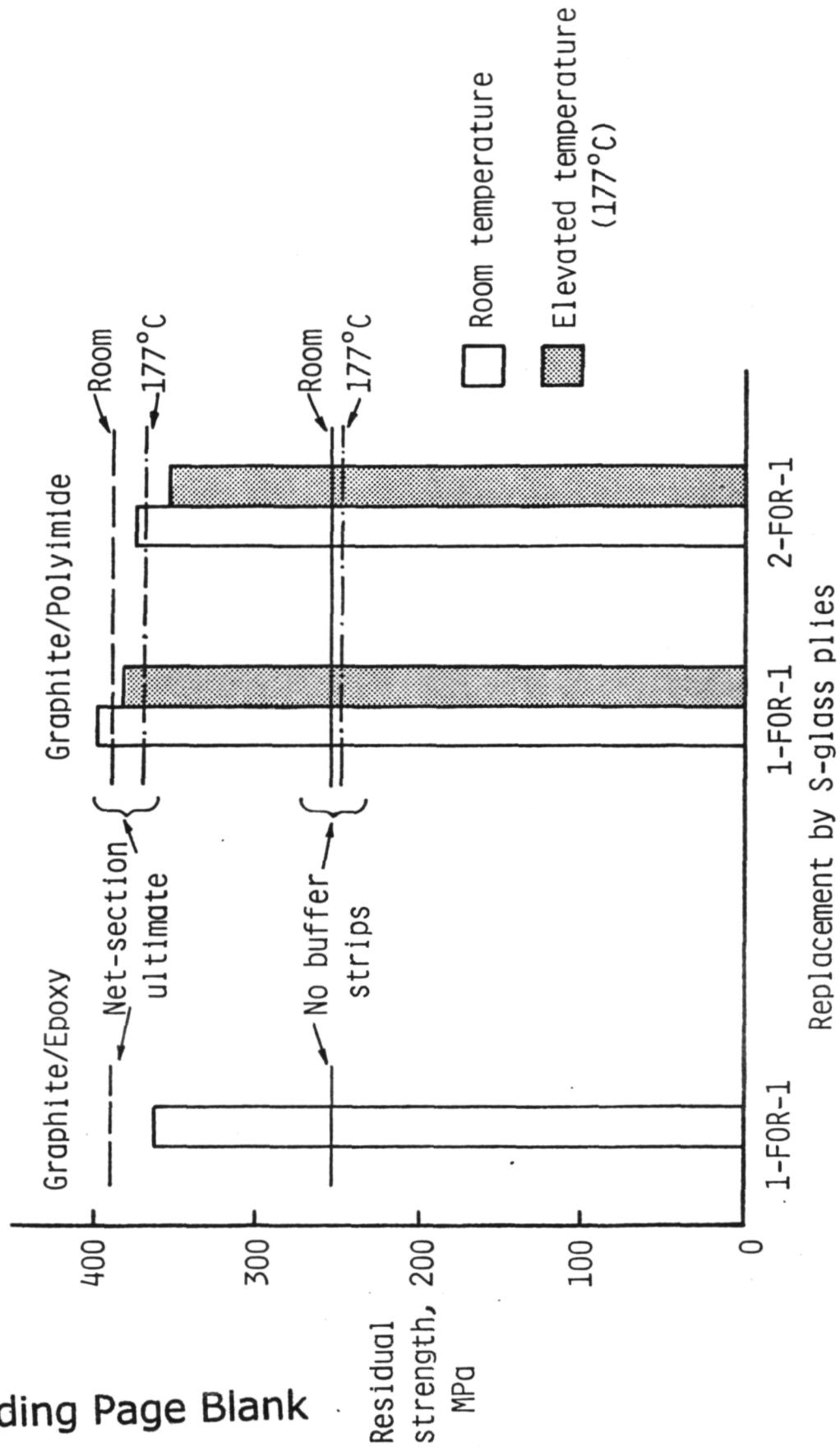
At the elevated temperature, the buffer strips increased the panel strength by at least 40 percent compared to panels without buffer strips. The buffer strip panels tested at 177°C exhibited stable crack growth until the crack grew into the buffer strip. Compared to similar panels tested at room temperature, the buffer strip panels tested at 177°C had lower residual strengths but higher failure strains. The panels with two-for-one buffer strip replacement had slightly lower strengths (by 5 percent) than panels with one-for-one replacement.

The figure shows the strength of the graphite/polyimide buffer strip panels. For comparison, the estimated residual strength is shown for a panel without buffer strips. Also, results for a graphite/epoxy buffer strip panel having the same layup and configuration are shown.

These static results have led to the next study in which graphite/epoxy buffer strip panels will be evaluated under fatigue loadings.

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Strengths of panels with buffer strips.

FRACTURE BEHAVIOR OF HYBRID COMPOSITE LAMINATES

Project Engineer: John M. Kennedy

An experimental program was undertaken to investigate the tensile fracture behavior hybrid laminates using center-notched fracture specimens studied. Three basic laminate groups were tested. The baseline group was all Thorne 300/Narmco 5208 (T300/5208) graphite/epoxy with laminate stacking sequences of $[45/0/-45/90]_{2S}$ and $[45/0/-45/0]_{2S}$. The second group had the same sequences, but the 0° plies were one or two layers of S-glass or Kevlar. The third group was all graphite/epoxy, but with perforated Mylar separating the 0° plies from other orientations; the laminate stacking sequences were $[45//0_N// -45/90]_{2S}$ and $[45//0_N// -45//0_N//45//0_N// -45//0_N]_S$, where $//$ indicates that the interface contains a layer of perforated Mylar and N was one or two. During tests, the load was increased monotonically while remote strain and crack-opening displacement (COD) were monitored. The notched region was radiographed periodically to reveal the extent and type of damage. A comparison of the radiographs show that the size and type of damage change substantially for the laminates considered. First, by changing the 0° plies from graphite to Kevlar or S-glass, the damage region became much larger because of the high failing strains of the Kevlar and S-glass fibers. Second, all of the laminates with Kevlar or S-glass plies split axially, whereas the $[45/0/-45/90]_{2S}$ all graphite laminate did not. Only when the 0° plies in $[45/0/-45/90]_{2S}$ laminate were isolated from the other plies using Mylar did axial splitting occur. Finally, increasing the percentage of 0° plies in a laminate, either by changing the fundamental laminate configuration or by adding additional 0° plies, produced longer axial splits.

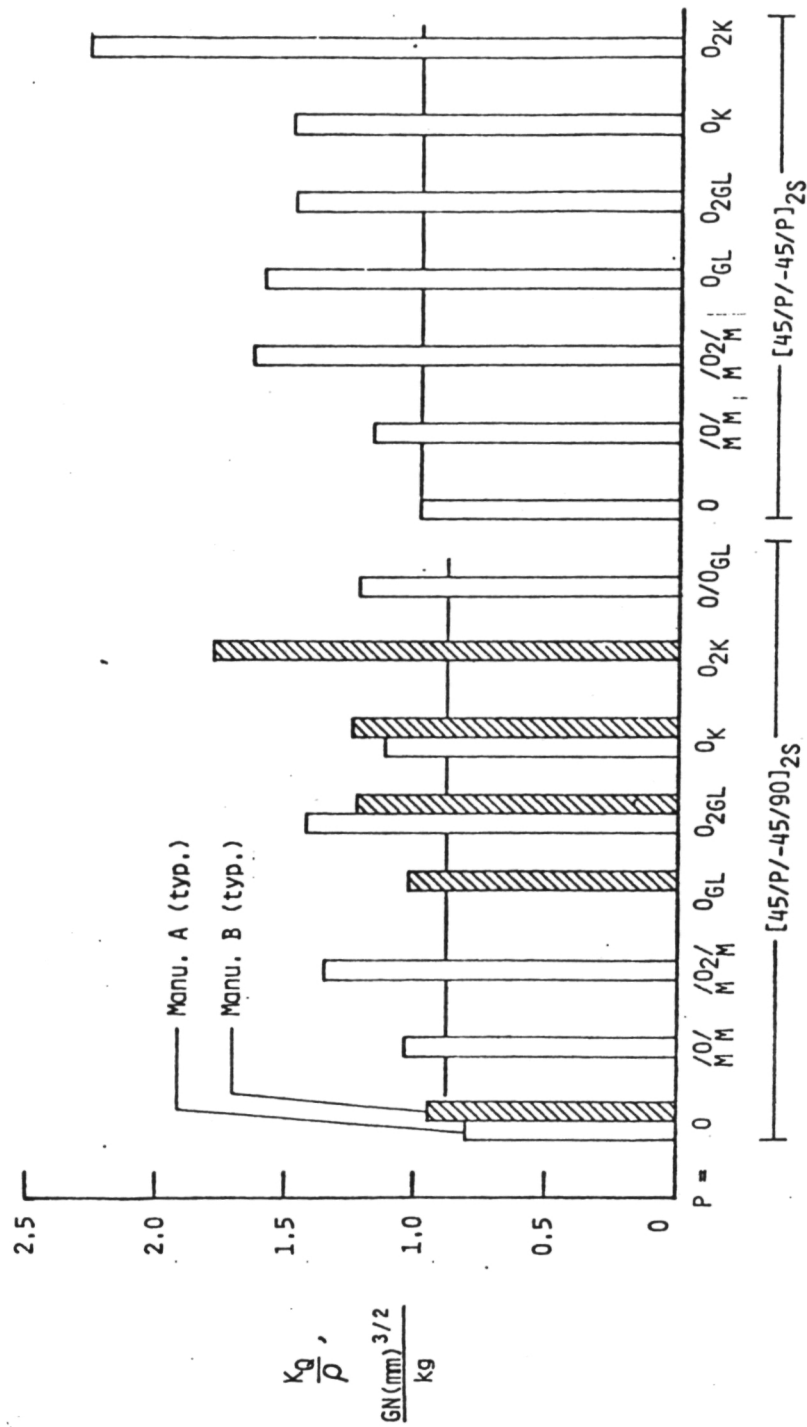
The structural efficiency for all of the laminates is presented in the figure in terms of a laminate's stress-intensity factor normalized with respect to its density. As the figure shows, all of the hybrid laminates are more efficient in terms of fracture than the all-graphite laminates. In general, the laminates with two adjacent 0° plies had higher K/ρ values than laminates with no adjacent plies.

The $K_{Q\rho}$ values plotted in the figure, along with the radiographs, show that, in general, the laminates with the most damage at the ends of the slit have the highest fracture toughness. Further, the $K_{Q/\rho}$ values of laminates with and without Mylar definitely shows that uncoupling the 0° plies from adjacent plies lowers the stress concentration at the end of the slit, which increases the toughness. The high strain S-glass and Kevlar plies produce a similar effect because the matrix fails before the Kevlar or S-glass fibers fail.

A general linear elastic fracture mechanics analysis for composites was used to predict the failing strains of the laminates. In general, the analysis underestimated the failing strain. This was expected since the analysis assumed a linear elastic response and did not account for damage at the ends of the slit.

Additional fracture tests are planned for these same hybrid laminates except the matrix will be changed to BP907 which has a high failing strain. The high strain matrix may not split and delaminate as much as the 5208 matrix, and consequently the fracture properties of the hybrid laminates may change.

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STRUCTURAL EFFICIENCY, COMPUTED FROM EXPERIMENTAL RESULTS,
OF ALL LAMINATES

DAMAGE TOLERANCE OF WOVEN GRAPHITE/EPOXY BUFFER STRIP PANELS

Project Engineer: John M. Kennedy

Previous tension tests on composite panels have shown that damaged panels with buffer strips had much higher residual strengths than panels without buffer strips. The panels were made from prepreg tape; the buffer strips were made by replacing narrow strips of the 0° graphite plies with either 0° S-glass/epoxy or 0° Kevlar/epoxy. The fabrication cost was very high, so technology was developed to make a less expensive buffer strip panel from woven cloth. The buffer material was woven into the graphite cloth. Panels of woven material were tested in tension and shear.

The woven cloth was a unidirectional weave with about 95 percent of the fibers in the warp direction. During tests, the load was increased monotonically while remote strain, and crack-opening displacement were monitored. The notched region and nearby buffer strips were radiographed periodically to reveal crack growth and damage.

The figure shows test results for tension buffer strip panels made from prepreg tape and woven cloth. The geometry of the panel is also shown. For all of the data shown, the buffer strips had one-ply thickness of 0° S-glass for each 0° ply of graphite. Panels were tested with machined notches 13 mm, 26 mm, and 44 mm long. As the load was increased, the crack ran to the buffer strip and arrested, and upon additional loading the panels failed. As the figure shows, the prepreg and woven panels have about the same failing strain or strength.

The figure also shows that the strain at which the cracks ran and the failing strain of the 48-ply panels were lower than those for the 16-ply panels. The differences in strains are due to differences in the amount of noncritical damage at the end of the notch or in the buffer strip.

The test results from the shear panels were similar to the results from the tension panels. The buffer strips arrested the crack and the failing strengths were higher than those of a panel with no buffer strips and similar damage.

Several analyses are available to predict the strength of buffer strip panels; however, only a shear-lag analysis accounts for buffer strip geometry and notch tip damage. Such an analysis correctly predicts the effects of buffer strip geometry and correlates with the experimental data.

This group of tests essentially completes the testing of buffer strip panels. The remaining task is to refine the shear-lag analysis to account for material and geometric parameters which can vary in a buffer strip panel.

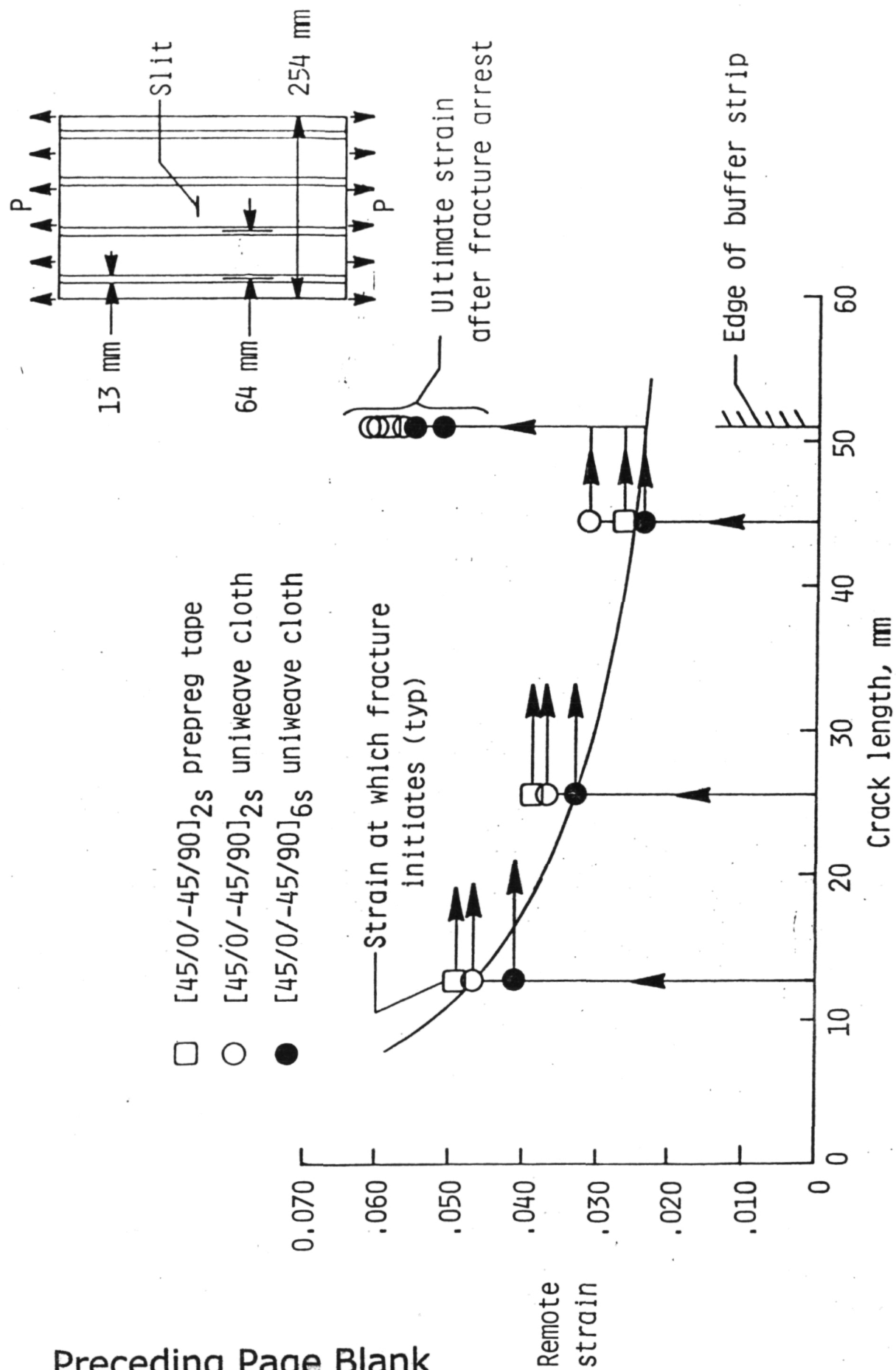


Figure 1.- Typical fracture arrest results for S-glass buffer strip panels.

SINGULAR STRESSES IN LAYERED ELASTIC BODIES WITH INTERFACE FLAWS

Project Engineer: John M. Kennedy

For a doctoral dissertation the behavior of perfectly bonded elastic bodies with disbonds on the bond line is being analyzed. This problem represents an abstraction of part of the impact problem in composites. The layer is loaded with a compressive normal load. Two types of disbonds were considered. The first is frictionless (no shear stress) and normal displacement differences between the layer and the half plane are zero, thus both tensile and compressive normal stress are allowed. The second type of disbond is frictionless also but normal displacement differences are zero only over a region where the normal stress is compressive. The disbond separates everywhere else.

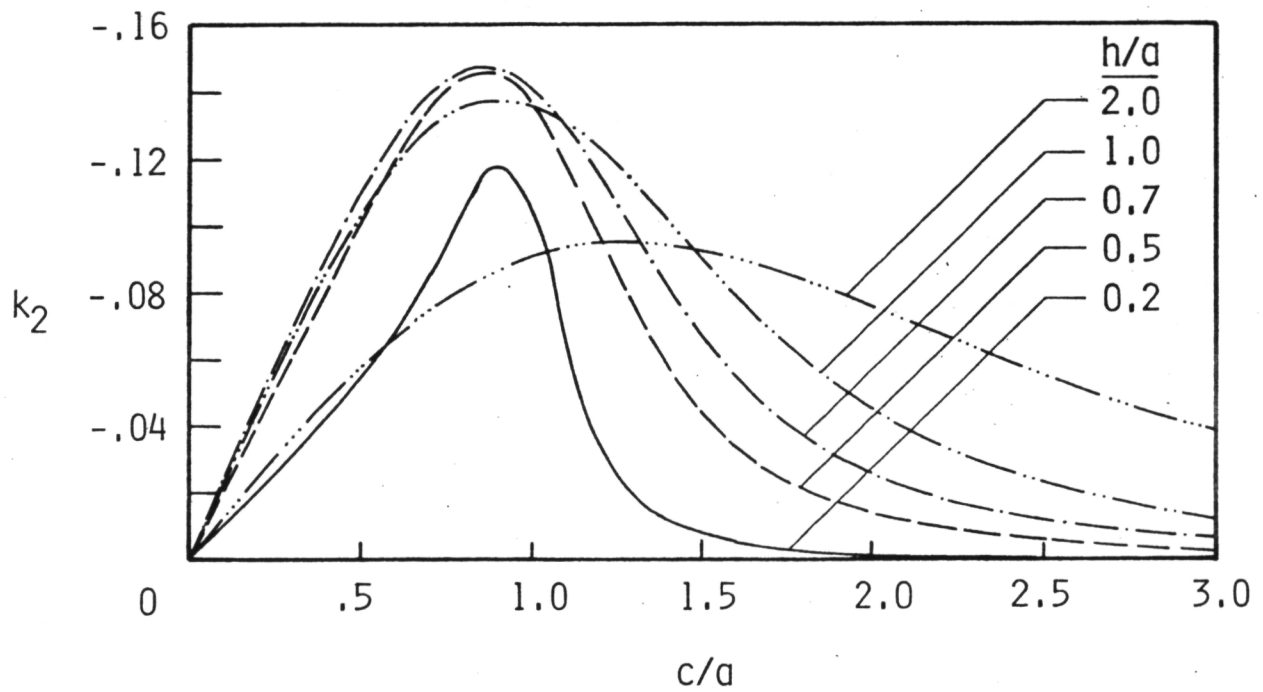
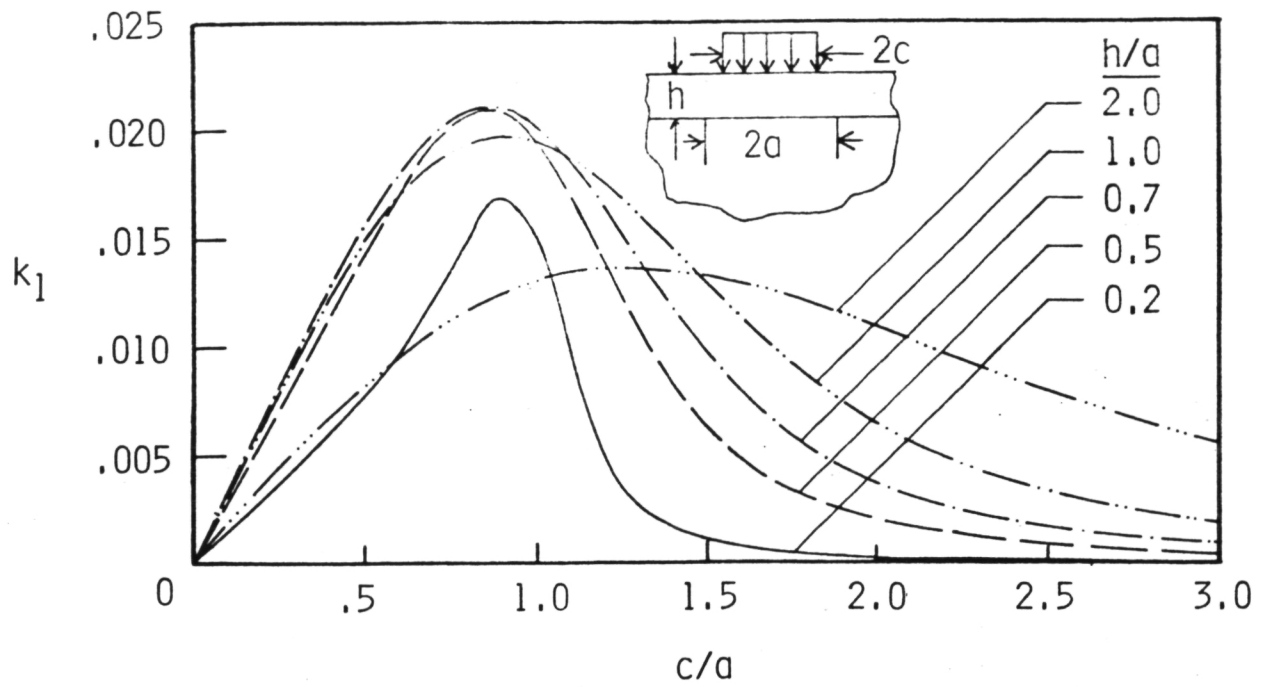
The governing equation for the first problem is a singular integral equation of the first kind and is readily solved numerically. The second problem is governed by a pair of coupled singular integral equations of the second kind and a constraint equation on normal displacement. The solution is obtained numerically in the least-squares sense because the equations are ill-conditioned.

The attached figure shows the stress intensity factors K_1 and K_2 plotted against applied load for various thicknesses of the layer. The disbond is the first type. The layer has aluminum elastic properties and the half-plane has steel elastic properties. The figure shows that both K_1 and K_2 are largest for a load width approximately equal to the disbond length; and significantly, K_1 is positive indicating a tensile stress at the end of the disbond. Also K_1 and K_2 are increasing with layer thickness up to $h = 7$ at which point they start to decrease.

Results for the problem with the second type of disbond are very similar. Of course, there is separation over approximately the same region where there were tensile stresses in the first problem. Trends in K_1 and K_2 are the same over the region where solutions were obtainable.

Many additional problems of this nature have not been solved, but there are no plans to continue the work.

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STRESS INTENSITY FACTORS FOR THE FULL CONTACT DISBOND

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ADHESIVELY LAMINATED METALS

Project Engineer: W. S. Johnson

In metallic structures, the fracture toughness for thick sections is often significantly less than for thin sections. The concept of building up thick sections by adhesively laminating thin sheets has previously been applied to aluminum. Aramid fiber-reinforced aluminum laminates (ARALL) have been used to successfully demonstrate superior crack growth lives and higher toughnesses.

In an experimental study, we have shown that adhesively bonded titanium sheets (without fiber reinforcements) provide similar benefits. In a comparison at a total thickness of 1 cm, the laminated titanium plate had 40 percent higher toughness than the monolithic plate for a 62 mm crack. Also, the fatigue life of the laminated plate was an order of magnitude greater than for the monolithic plate for the same initial 5 mm flaw in one outer sheet.

Demonstration articles of laminated metal have been built and tested at General Dynamics/Fort Worth and LTV Aerospace and Defense. Research on laminated metals is currently under way at Alcoa, LTV, Fokker, and The Delft University.

Publication:

Johnson, W. S.: Damage Tolerance Evaluation of Adhesively Laminated Titanium. Journal of Engineering Materials and Technology, vol. 105, no. 3, July 1983, pp. 182-187.

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FATIGUE AND FRACTURE OF CONTINUOUS FIBER METAL MATRIX COMPOSITES

Project Engineer: W. S. Johnson

Metal matrix composites (MMC), in spite of their relatively high cost, have several inherent properties that make them attractive for structural applications: MMC have high stiffness-to-weight and strength-to-weight ratios and better transverse strength, operative temperature range, and environmental resistance than competitive epoxy-resin matrix composites.

Two MMC systems have been studied for fatigue behavior: boron/aluminum and silicon-carbide (SiC_2)/aluminum. Both systems can develop significant internal matrix cracking when fatigued. These matrix cracks can result in a 40 percent secant modulus loss in some laminates, even when fatigued below the fatigue limit. A simple analysis that predicts unnotched laminate secant modulus loss due to fatigue has been developed. The analysis is based upon the elastic modulus and Poisson's ratio of the fiber and matrix, fiber volume fraction, fiber orientations, and the cyclic-hardened yield stress of the matrix material. Excellent agreement was achieved between model predictions and experimental results. With this model, designers can project the material stiffness loss for design load or strain levels and assess the feasibility of its use in stiffness-critical parts. This work was cited as the approach to use for studying fatigue response of MMC in the Air Force's "Durability of Continuous Fiber-Reinforced Metal Matrix Composites."

Boron/aluminum laminates were tested to enhance the understanding of the fracture process of MMC. This work was performed with Cathy Bigelow and is reported separately. MMC can experience widespread yielding of the matrix prior to the fracture process. Specimen stress-strain behavior, stress at first fiber failure, and ultimate strength were experimentally determined for five different layups. Radiographs were used to monitor the fracture process. For notched unidirectional specimens, the first fiber failure occurred at approximately one-half of the specimen ultimate strength. The first fiber failures in notched specimens with $[\pm 45]_{2S}$, $[0/\pm 45]_S$, and $[0_2/\pm 45]_S$ laminate orientations occurred at or very near the specimen ultimate strength. Circular holes and crack-like slits of the same characteristic length were found to produce approximately the same strength reductions.

MMC research is currently under way at Lockheed-Georgia Company, McDonnell-Douglas, AMMRC, and LTV Aerospace and Defense.

Publications:

1. Johnson, W. S.: Mechanisms of Fatigue Damage in Boron/Aluminum Composites. Damage in Composite Materials, ASTM STP 775, 1982, pp. 83-102.
2. Johnson, W. S.: Modeling Stiffness Loss in Boron/Aluminum Laminates Below the Fatigue Limit. Long-Term Behavior of Composites, ASTM STP 813, 1983, pp. 160-176.
3. Johnson, W. S.; Bigelow, C. A.; and Bahei-El-Din, Y. A.: Experimental and Analytical Investigation of the Fracture Processes of Boron/Aluminum Laminates Containing Notches. NASA TP-2187, September 1983.

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ANALYTICAL AND EXPERIMENTAL INVESTIGATION OF THE FRACTURE PROCESSES OF
BORON/ALUMINUM LAMINATES CONTAINING NOTCHES

Project Engineer: C. A. Bigelow

The purpose of this study, carried out jointly with W. S. Johnson, is to enhance the understanding of the fracture processes of metal matrix composites--boron/aluminum composites, specifically. These composites experience widespread yielding of the matrix material during the fracture process. Such plasticity complicates the analysis of composites containing discontinuities, such as crack-like slits or circular holes. Experimental results have been obtained for five laminate orientations of boron/aluminum composites containing either crack-like slits or circular holes. Specimen stress-strain behavior, stress at first fiber failure, and ultimate strength were determined. Radiographs were used to monitor the fracture process. The configurations were analyzed with a three-dimensional elastic-plastic finite-element program. The program uses a continuum material model with an eight-noded, hexahedral element. The program calculates matrix stresses and strains, and fiber stresses. It can model the nonlinear behavior of metal matrix composites and predict the stress level of first fiber failure.

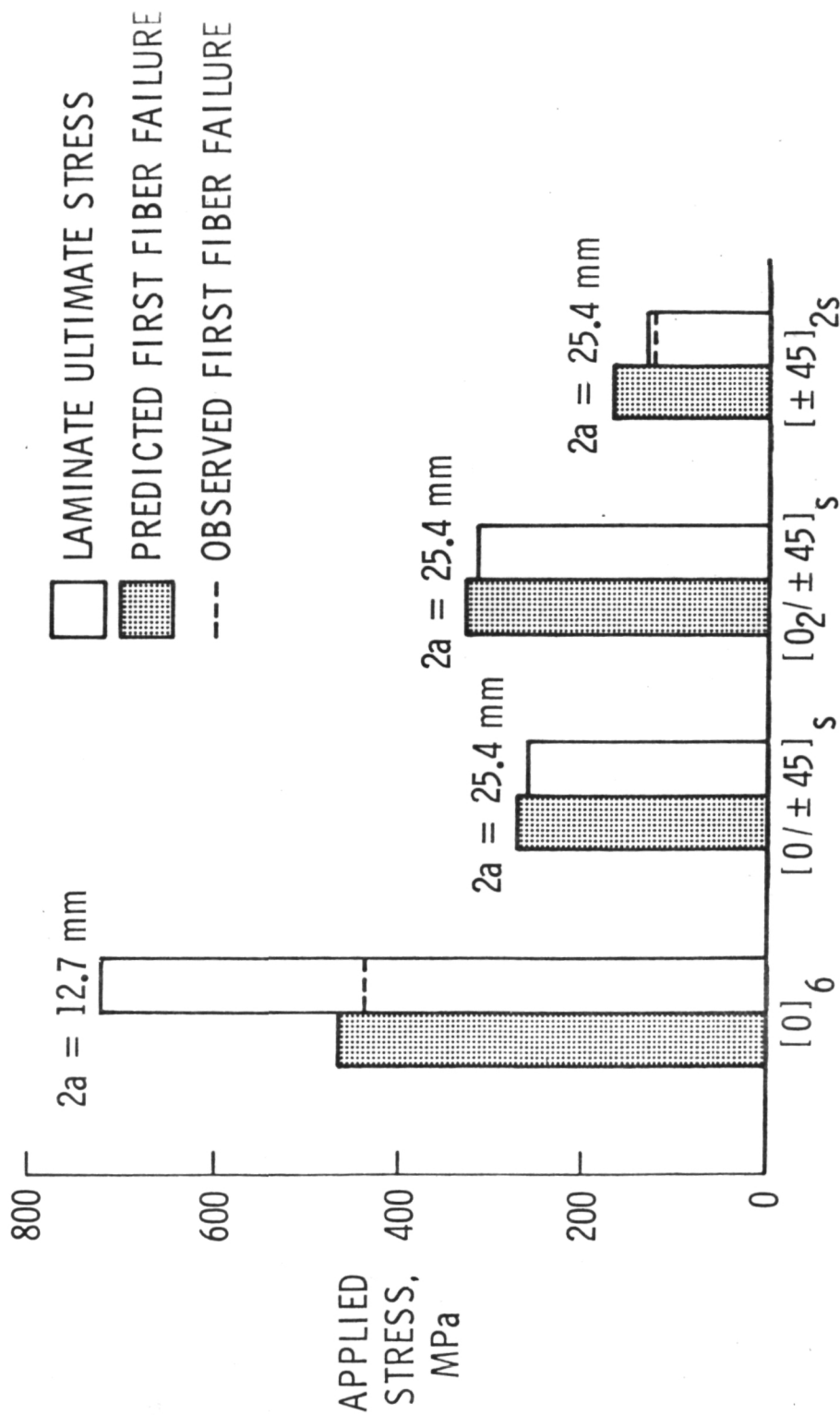
The first fiber failure in notched specimens with $[\pm 45]_{2S}$, $[0/\pm 45]_S$, and $[0_2/\pm 45]_S$ laminate orientations occurred at or very near the specimen ultimate strength. For notched unidirectional specimens, the first fiber failure occurred at approximately one-half of the specimen ultimate strength, as shown in the figure. Acoustic emission events correlated with fiber breaks in unidirectional laminates but did not for other laminates. Circular holes and crack-like slits with the same characteristic length had about the same strength reduction. The predicted stress-strain responses and stress at first fiber failure compared very well with test data for laminates containing 0° fibers and reasonably well for $[\pm 45]_{2S}$ laminates.

These results are significant since, with this finite-element program, fracture behavior of metal matrix composite laminates can be predicted. The program can model the nonlinear behavior of the composite and predict the point of first fiber failure.

Additional verification of the model predictions is still in progress.

Publication:

Johnson, W. S.; Bigelow, C. A.; and Bahei-El-Din, Y. A.: Experimental and Analytical Investigation of the Fracture Processes of Boron/Aluminum Laminates Containing Notches. NASA TP-2187, September 1983.



COMPARISON OF ULTIMATE STRENGTH AND FIRST FIBER FAILURE
FOR LAMINATES WITH CIRCULAR HOLES

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16. Abstract This compendium of recently completed and ongoing research projects from the Fatigue and Fracture Branch at NASA Langley Research Center provides technical descriptions and key results of all such projects expected to lead to publication of significant findings. The common thread to all these studies is the application of fracture mechanics analyses to engineering problems in metals and composites, with particular emphasis on airframe structural materials. References to recent publications are included where appropriate.					
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